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GEORGE C. MARSHALL SPACE FLIGHT CENTER

HUNTSVILLE, ALABAMA

LUNAR LOGISTIC SYSTEM

VOLUME VI

TRACKING AND MISSION CONTROL

BY

AEROBALLISTICS DIVISION

National Aeronautics and Space Administration

NASA

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By Flight Evaluation Branch,
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ABSTRACT

This volume presents results of the Lunar Logistic System studies in three related areas: Tracking and Orbit Determination, Midcourse Maneuver Requirements, and Mission Control. The principal conclusions derived in each area are given below. More detailed summaries of results are given after each individual primary division of the report.

Tracking and Orbit Determination:

Tracking of the logistics vehicle throughout the mission profile will primarily be performed by ground stations. The studies reported were restricted to the earth-moon transit and lunar orbit phases, where primary tracking was assumed by the NASA Deep Space Network.

Two types of errors enter the tracking and orbit determination process: random observational errors, generally amenable to statistical smoothing; and systematic errors in both observations and the mathematical trajectory model used for orbit determination. Systematic errors in the orbit determination are substantial, but planned instrumentation is sufficiently accurate to accomplish their reduction. Ranging capability is important in this regard.

The desired landing accuracy of 2 km can be achieved in the direct mode if range data is available, and without range data if a lunar beacon is available during the terminal descent. Horizontal approach trajectories are less accurate than perpendicular approaches.

In the lunar orbit mode, the parking orbit can be established within permissible limits. Range data is required for the desired accuracy of landing from orbit if a lunar beacon is not available. With a beacon, however, ranging is not required. Two to three revolutions in the parking orbit appear necessary.

Midcourse Maneuver Requirements:

The exact requirements and procedures for midcourse corrections will depend upon the final scheme adopted. A general survey of possible schemes and a few specific cases have been examined.

Two midcourse corrections should be prepared for, the first to be performed about 10 hr after injection and the second 30 to 50 hr after injection.

The midcourse ΔV requirement is essentially determined by the magnitude of injection guidance errors. The ΔV requirements for midcourse corrections and lunar orbit braking maneuver corrections are closely interrelated, and an optimization of the maneuver scheme considering both maneuvers appears desirable.

Lunar Logistic Mission Control:

A concept is presented to provide necessary mission control functions while making optimum use of existing facilities. Under this concept, existing and planned ground tracking stations would be used as well as an existing facility, designated the Ground Instrumentation Control Center, for control of the ground network. A separate but moderate scale Mission Control Center would perform overall mission and vehicle control. Since the mission control is tightly linked to the vehicle design, the Mission Control Center should be operated by the vehicle developer for efficient operation and reliable control.

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I. GENERAL INTRODUCTION

This volume summarizes the results of studies performed by the Flight Evaluation Branch of Aeroballistics Division in support of a Lunar Logistics System. The studies were in three related areas: Tracking and Orbit Determination, Midcourse Maneuver Requirements, and Mission Control.

The first topic concerns the means of measuring the vehicle flight path, and the accuracy of measurement. The second deals with the scheme for control of the vehicle earth-moon transit, and its influence on propellant requirements and orbit determination. Third and last topic presents a concept of the ground support complex required to exert operational control on the vehicle during flight.

II. TRACKING AND ORBIT DETERMINATION

A. INTRODUCTION

This section reports on studies of tracking accuracy for a soft lunar landing. Both direct and lunar orbit modes of landing have been considered for comparison purposes. The results indicate possible landing accuracies and some of the tracking instrumentation and profile tradeoffs to be weighed in planning the mission. Further and more exhaustive analysis is required to achieve definitive conclusions.

B. ASSUMPTIONS AND LIMITATIONS OF STUDY

The analysis considers the errors that may be expected for a logistic vehicle landing due to error in orbit determination alone. In general, the results should be regarded as order of magnitude estimates only. Several factors limit the accuracy of the results.

The analysis is for the most part based upon a least-squares orbit determination whose accuracy is estimated by linear error theory. However, a number of non-linear simulations of the orbit determination have been made for various error sources. The agreement between the linear and non-linear simulations permits use of the linearized results for system characteristics, but also reveals that a non-linear Monte Carlo analysis is required if precise results are desired.

The accuracy of the analysis is also limited by the restricted number of error sources considered and by the degree of realism of error assumptions concerning the tracking instrumentation to be used. Since the instrumentation involved is to a degree to be built in the future, the precise characteristics are unknown and estimates must be used. The error sources which should be incorporated and their importance depend both upon the instrumentation characteristics assumed and upon the success of flights prior to the logistic vehicle mission in reducing or eliminating certain errors.

The influence of midcourse corrections during the earth-moon transfer trajectory on terminal accuracy has been separately considered in Section III of this report.

It has been assumed that primary tracking will be performed by three stations of the existing Deep Space Instrumentation Facility (DSIF) or similar stations. Each station performs angle, range, and range rate measurements with the following 1σ random errors:

Angles $\pm$ 0.04 deg
Range Rate $\pm$ 0.2 m/s
Range $\pm$ 15 m

Uncorrelated measurements are assumed to be obtained at 10 second intervals. No data is used below five degrees elevation angle.

For some purposes, tracking of a beacon on the lunar surface from onboard the vehicle is considered. It is assumed that measurements of angles, range, and range rate can be performed with the following 1σ random errors:

Angles $\pm$ 0.3 deg
Range Rate $\pm$ 10 m/s
Range $\pm$ 15m

Uncorrelated measurements of beacon data are assumed at 1 second intervals. No data is used below five degrees elevation.

A number of systematic error sources affect an orbit determination. These sources fall into two categories: those that affect both the interpretation of tracking data and the prediction of the vehicle flight path, and those which affect only the tracking measurements or their interpretation. Table I shows a list of these systematic error sources. Those marked by an asterisk have been considered in the present investigation.

The systematic errors which have not been considered are by no means all negligible, but were omitted due to time limitations. For example, the lunar distance uncertainty contributes a significant error, almost as large as that of the moon mass. The systematic errors considered have all been regarded as independent; although significant correlations will exist between some of the astronomical constants due to the method of their measurement. This factor will complicate a more thorough analysis.

C. LINEARITY OF SYSTEMATIC ERROR EFFECTS

The validity of a linearized error analysis depends upon the degree of linearity of the transformation between the error source whose distribution is known and the mission parameter whose error is desired. Figure 1 illustrates the range of cases which may be encountered. The upper portion of the figure shows the difference between the actual and predicted lunar latitude of the direct mode landing points as a function of the error in the earth gravitation constant assumed in a non-linear orbit determination and prediction process. A reasonably linear relationship applies, and a linearized analysis based on a -1σ perturbation of GM yields reasonable agreement, as indicated by the dashed curve.

TABLE I

PHYSICAL CONSTANTS

Affecting Measurements	Present Uncertainty (1σ)
* Velocity of Light	$\pm$ 0.0001% (R)
* Station Coordinates (3 per Station)	$\pm$ 100 m
Systematic Instrument Errors	-
Atmospheric Refraction	-
Affecting Orbit Determination and Prediction	
* Earth Gravitation Constant (GM)	$\pm$ 0.001% (R,J)
* Moon Mass/Earth Mass	$\pm$ 0.03% (R)
* Lunar Oblateness	$\pm$ 10% (B)
* Lunar Equatorial Ellipticity	$\pm$ 25% (B)
Lunar Position (3 Coordinates)	$\pm$ 2 km (B)
Small Perturbations	-

* Considered in Present Study
B Baker, Makemson, and Westrom
J JPL (Clarke)
R Rand Memo 2944 (de Vaucouleurs)

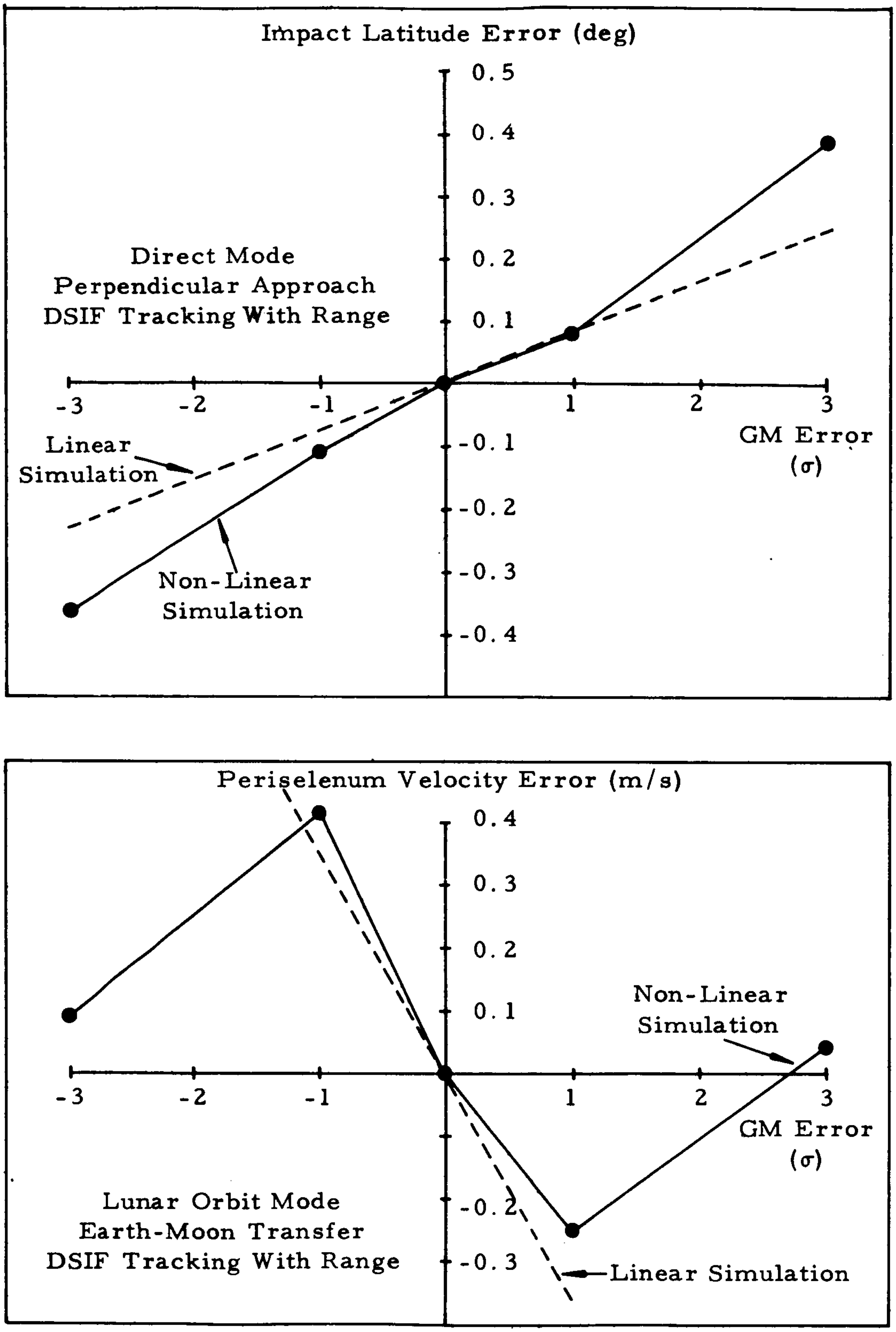


FIG. 1 LINEARITY OF GM ERROR

On the other hand, the lower portion of Figure 1 shows the difference in actual and predicted velocity of an earth-moon transfer trajectory at the time when injection into a lunar parking orbit might take place. The velocity error is again shown as a function of error in the GM used in the orbit determination process. In this case, the linearized result (dashed curve) does not represent the true picture.

Both of the examples given assumed DSIF tracking with range data during essentially the entire earth-moon transfer.

When the error transformation is as non-linear as in the last example given, a Monte Carlo type process should be used to derive the true maximum error to be expected at a given confidence level. A complete non-linear analysis of this type has not been performed in the present study. The procedure followed as far as possible has been to derive through non-linear simulation the mission errors due to specific systematic errors (e.g., $\delta GM = \pm 1\sigma$, $\pm 3\sigma$) and compare these mission errors with those predicted by the linearized technique. It has not been possible to perform non-linear simulations of all error sources; so that in general the linearized results are shown, suitably adjusted where disagreement with non-linear simulations is found.

In general, most difficulty with the linear model is encountered for treatment of those error sources affecting both the orbit determination and prediction; in this case, compensation is experienced between the error committed in the orbit determination and that in the orbit prediction. Although the separate errors are large, the resultant total error is quite small.

D. DIRECT MODE LANDING ERROR

The direct mode was analyzed for a scheme which did not utilize either a beacon on the lunar surface or other lateral position information during the terminal landing phase. It is assumed that an orbit determination will be made some time before the vehicle reaches the moon. The time at which the vehicle will reach 500 km lunar altitude is predicted, and a timer is set to command engine ignition at this time. A powered descent to a soft landing is then made inertially with initial position and velocity information as predicted by the midcourse orbit determination. The landing errors derived under this scheme might be reduced if terminal position information is available during the powered descent. On the other hand, inevitable small errors of the inertial guidance are not accounted for and will tend to increase the pure tracking errors shown.

The landing error on the lunar surface is composed of error in two coordinates, so that the error distribution is two-dimensional. Under the linearized error theory, the distribution is normal bivariate and the confidence contours of the distribution are given by ellipses on the surface. For all cases of practical interest the deviation in time of arrival is insignificant for the landing scheme, since an on-board altimeter may be assumed to be employed during the terminal phase to remove altitude error. Only the radial distance error on the surface of the nominal point is considered here, and is characterized by the semimajor axis of the confidence ellipse containing 99.5% of the landing errors.

1. Perpendicular Approach Trajectory. A 68-hour perpendicular approach trajectory is assumed tracked by the DSIF from shortly after earth injection until about five hours before moon landing. Tracking after this time is not considered for navigation purposes, in order to allow ample time for orbit determination and transmission of necessary commands to the vehicle. A slight additional gain in landing accuracy can be obtained if this end time is relaxed (see Par. IIE). This result may be altered if a midcourse correction is performed late in the trajectory (see Par. IIIF).

In Figure 2 the landing error due to error in the assumed gravitational constant of the earth and the mass of the moon is shown as derived by both linear and non-linear simulation. Two cases are considered, for tracking with and without DSIF ranging data. For each of the two physical constants considered, landing error is derived by non-linear simulation of constant errors of $\pm 1\sigma$ and $\pm 3\sigma$. The landing error from each simulation is adjusted to represent the equivalent semimajor axis of the 99.5% confidence ellipse if the error transformation relationship were linear. If the relationship were linear, the adjusted landing errors shown for the various simulations of each error source would be equal and would stay precisely on vertical lines. The degree to which they differ with each other and with the linearized simulation result is indicative of the validity of the linear assumption.

The solid points represent landing errors if a hard impact is considered, while the circles represent landing errors if a powered descent, as described previously, is considered.

In all cases, the linearized hard impact error agrees well with the hard impact errors derived by non-linear simulation. In all cases except that of a GM error with range tracking, there is little difference between the non-linear hard impact and soft landing errors. In the excepted case, the soft landing error exhibits considerable non-linear variation with the magnitude of the GM error, and is on the average

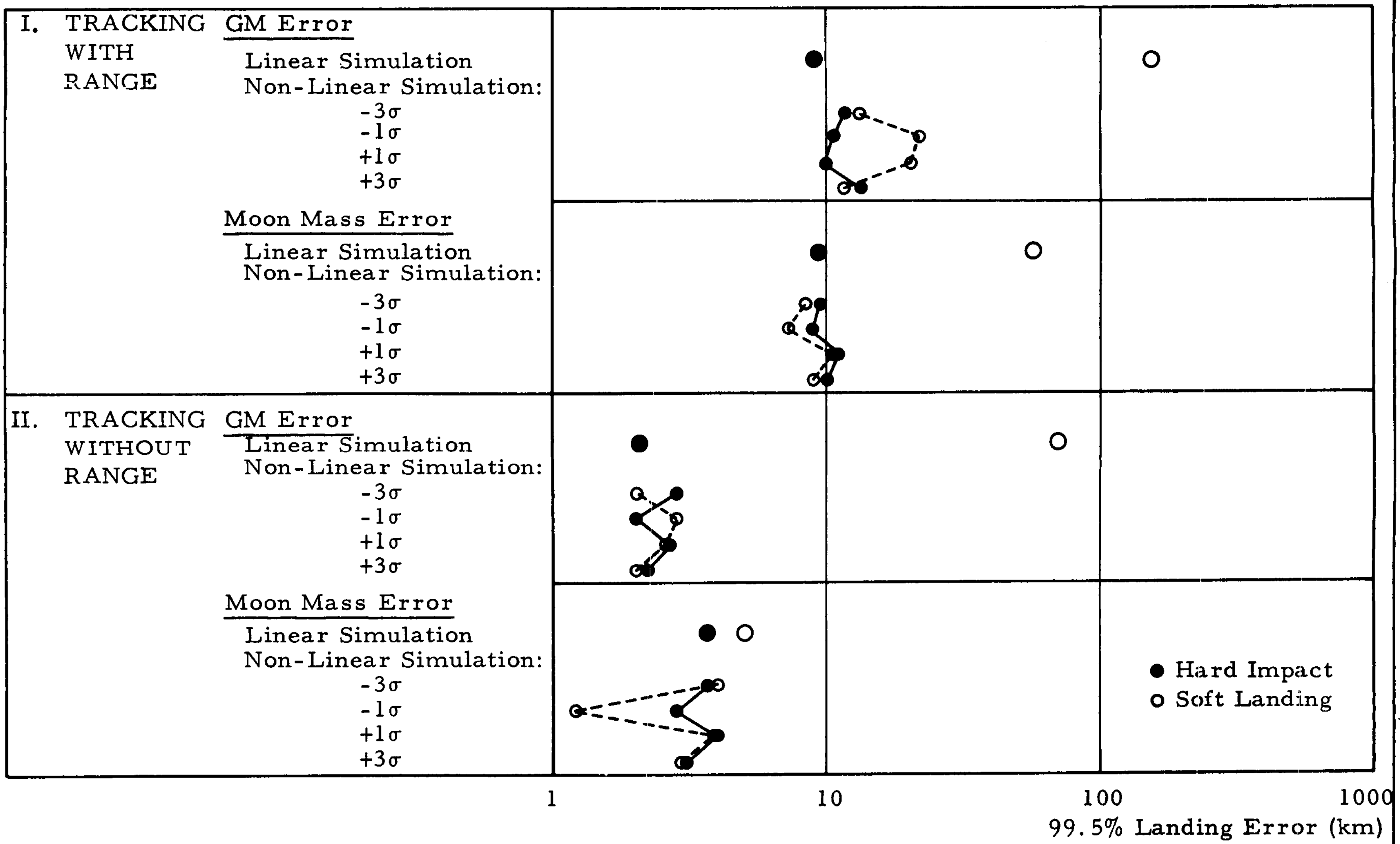


FIG. 2 LINEAR AND NON-LINEAR SIMULATION COMPARISON PERPENDICULAR DIRECT APPROACH TRAJECTORY

somewhat larger than the hard impact error. However, in all cases, the linearized simulation method breaks down for the soft landing and fails to represent the true error. The principal area of non-linearity is in representing the errors at the fixed time of power flight initiation.

The principal conclusion from the preceding results is that the linearized landing errors derived for a hard impact may be utilized to generally represent the true errors to be expected in a soft landing trajectory.

The systematic errors previously enumerated are the dominant error sources. The error contributions of the various constant uncertainties and the random observational error are shown in Table II. These and the following results for a perpendicular approach trajectory were derived by linearized analysis of a hard impact trajectory, but are to be interpreted as equivalent to soft landing errors as just shown.

In Table II it will be seen that the systematic errors are significantly smaller without range data than with range data. However, the random observational error is much larger without ranging; which also means the potential landing accuracy (Item IV in Table II), if the systematic errors are removed by treatment as unknowns in the orbit determination, is much better with ranging. This potential accuracy is that theoretically obtained by solving for the systematic error sources simultaneously with the state variables in the orbit determination. However, achievement of this potential accuracy raises many problems, particularly if required in real time during a flight. It must also be noted that additional systematic errors not considered here will become significant contributions to the total error at the low potential error level with range tracking.

Figure 3 further depicts the landing error relationship between systematic and random sources, with and without ranging. The total landing error for both the systematic and random sources considered is shown as a function of the present uncertainty of the physical constants. Our present knowledge is represented by 100%. The landing error will be reduced as the constant uncertainties are reduced. The potential gain without range data is small, however, due to the random noise level of the range rate data. With ranging, landing errors of less than 1 km are possible.

TABLE II

PERPENDICULAR APPROACH LANDING ERRORS

	Error Sources	99.5% Landing Error (km)	
		DSIF With Range	DSIF Without Range
I.	Random Observational Only	0.04	3
II.	Systematic Errors:		
	Station Coordinates	12	5
	Earth Gravitation Constant	9	2
	Moon Mass	10	4
	Velocity of Light	5	0.5
III.	Total Systematic and Random Observational	18	7
IV.	Theoretical Error When Solving for Systematic Errors	0.3	4

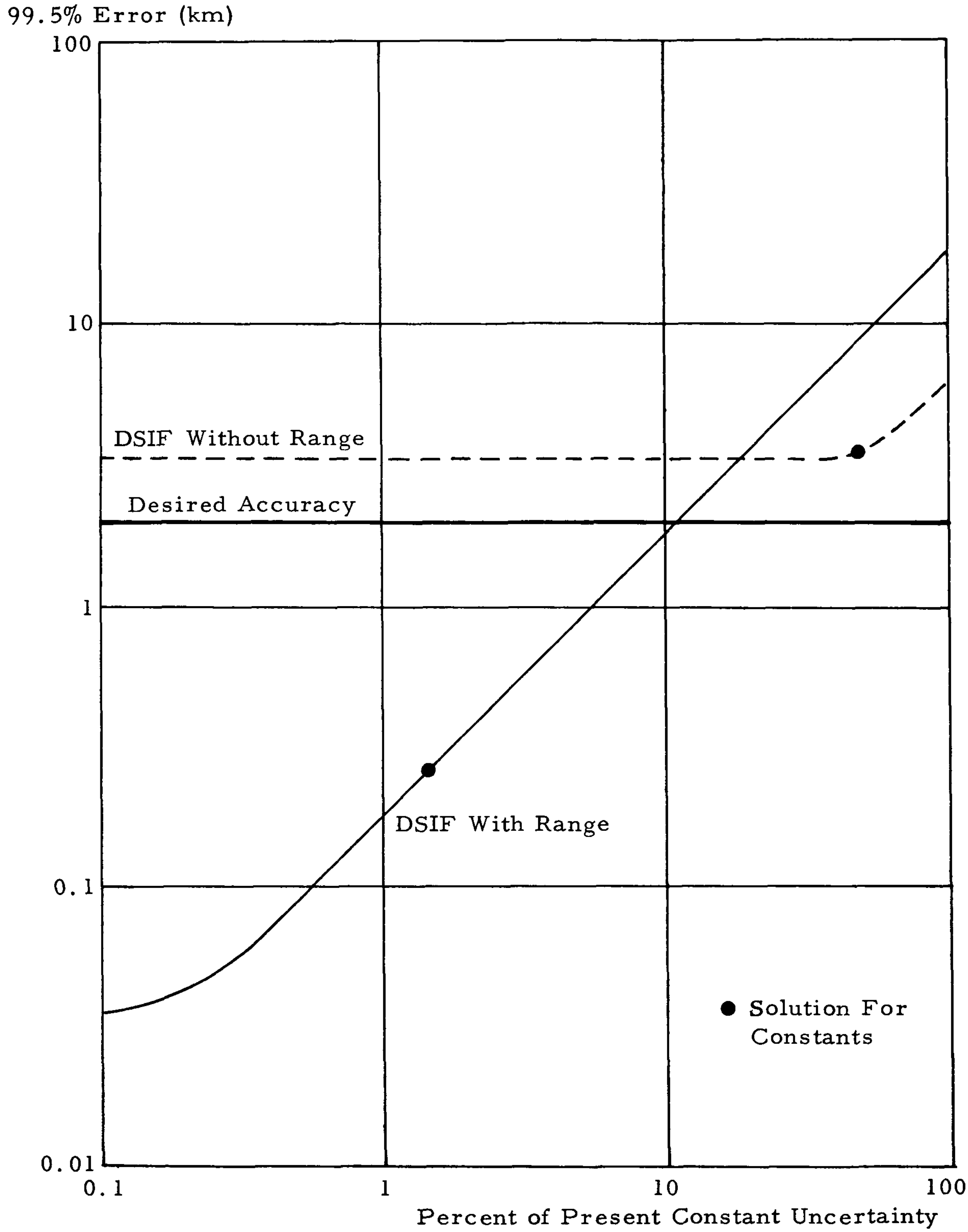


FIG. 3 DIRECT MODE LANDING ERROR PERPENDICULAR APPROACH TRAJECTORY

2. Horizontal Approach Trajectory. The near agreement between hard impact and soft landing errors observed for the perpendicular approach trajectory does not apply for a near horizontal approach trajectory. The trajectory considered here was a 66-hour transfer trajectory with a velocity vector 30 degrees below the local lunar horizontal at 500 km lunar altitude and essentially horizontal at hard impact. Linearized analysis of systematic errors for a hard impact indicated errors of a few hundred kilometers, while non-linear simulations of the same error sources revealed that in many cases no hard impact occurred, but the trajectory rather missed the moon with a low periselenum altitude. Of course, hard horizontal approach impact errors are of little interest.

Table III shows the landing errors due to GM and moon mass errors for soft landing for both the horizontal and perpendicular approach trajectories and the effect due to linearization. The linearized results do not agree with the non-linear simulations for either trajectory. The non-linear simulation results are again adjusted such that they would be constant if the errors were linear. Note that the disagreement here is by an inconsistent factor in each separate comparison. The non-linear simulation results indicate that the landing error due to GM error is about the same for both trajectories, while the landing error due to moon mass error is about four times larger for the horizontal approach than for the perpendicular approach. This is true for tracking with and without range.

Table IV shows a comparison between the hard impact error for the perpendicular approach trajectory (equivalent to soft landing error) and the horizontal approach soft landing error as derived by linearized simulation (except in the cases of GM and moon mass), where the non-linear simulation error is given. There is some evidence to support the linearized results shown for the horizontal approach trajectory although the linearized results have previously been shown erroneous for the GM and moon mass errors. As noted before, difficulty with the linear simulation was encountered principally with those systematic errors which affect the trajectory prediction model (GM and moon mass). Further, the comparison in Table IV between perpendicular and horizontal approach accuracies shows horizontal approach errors larger by a roughly consistent factor of about three for both large and small errors, as opposed to the more gross and random discrepancies noted in Table III and elsewhere when the linearized simulation breaks down.

Although these results are to a degree inconclusive, horizontal approach appears less accurate than perpendicular approach by a factor of about three to four.

TABLE III

EFFECT OF LINEARIZATION ON SOFT LANDING ERRORS

		99.5% Landing Error (km)	
		Perpendicular Approach	Horizontal Approach
I. With Range Tracking			
GM Error			
Linearized		154	91
Simulation:	-3σ	13	11
	-1σ	24	7
	$+1\sigma$	21	46
	$+3\sigma$	12	68
Moon Mass Error			
Linearized		59	9
Simulation:	-3σ	9	35
	-1σ	7	31
	$+1\sigma$	11	30
	$+3\sigma$	9	26
II. Without Range Tracking			
GM Error			
Linearized		68	63
Simulation:	-3σ	2	3
	-1σ	3	2
	$+1\sigma$	3	5
	$+3\sigma$	2	5
Moon Mass Error			
Linearized		5	6
Simulation:	-3σ	4	9
	-1σ	1	10
	$+1\sigma$	4	5
	$+3\sigma$	3	8

TABLE IV

COMPARISON OF SOFT LUNAR LANDING ERRORS

Error Source		99.5% Landing Error (km) Perpendicular Approach	99.5% Landing Error (km) Horizontal Approach
I. With Range Tracking			
A.	Random Observational	0.04	0.13
B.	Systematic Errors		
	Station Coordinates	12	33
	Earth Gravitation Constant	9	68*
	Moon Mass	10	35*
	Velocity of Light	5	12
C.	Total Systematic and Random Observational	18	84*
D.	Theoretical Error When Solving For Constants	0.3	1
II. Without Range Tracking			
A.	Random Observational	3	9
B.	Systematic Errors		
	Station Coordinates	5	24
	Earth Gravitation Constant	2	5*
	Moon Mass	4	10*
	Velocity of Light	0.5	3
C.	Total Systematic and Random Observational	7	28*
D.	Theoretical Error When Solving For Constants	4	11

* Adjusted to agree with non-linear simulation

It has been assumed for the horizontal approach that tracking for navigational purposes is ceased five hours prior to lunar landing, similarly as for the perpendicular approach. Reducing this time restriction to about three hours before landing yields little increase in accuracy, but reduction to one hour before landing would improve the accuracy by a factor of two. The effect of a late midcourse maneuver would alter these results.

E. LUNAR ORBIT MODE

1. Lunar Parking Orbit Establishment. An essential step in landing by the lunar orbit mode is establishment of a lunar parking orbit. The periselenum of the parking orbit was selected as the most critical parameter in its establishment, all other elements appearing to be less important within reasonable limits. The primary concern is that the periselenum lies reasonably above the surface of the moon.

The linear and non-linear simulations of GM and moon mass errors are compared in Figure 4. The non-linear simulation results are again adjusted so that they would be constant if the error relationship was linear. Strong non-linear behavior is observed and the linear method generally overestimates the periselenum errors, in one case by an order of magnitude. The contributions of various error sources to the 3σ periselenum error is given in Table V. Errors as derived from linearized simulation are given except as noted in the table.

Similarly as in the direct mode, the systematic errors with range data are larger than those without. The random observational error is two orders of magnitude larger without ranging, however, and the potential accuracy through simultaneous solution for state variables and systematic errors is a factor of four better with range than without.

The potential improvement of the periselenum accuracy with reduction of constant uncertainties is shown in Figure 5. The desired accuracy can be achieved without ranging data if systematic errors can be reduced to about 70% of their present values. However, much better accuracies can be achieved with ranging. The theoretical accuracies that can be achieved through solution for the constants are indicated by the solid circles. It must be remembered, however, that additional systematic errors not considered here will become significant as the total error level is reduced.

The results shown are, as in the direct mode, based on the conservative assumption that the final orbit determination begins five hours before time for braking maneuver into lunar orbit, in order to allow ample time for the computation and command transmission sequence. No tracking data were used after this time. Figure 6 shows the effect

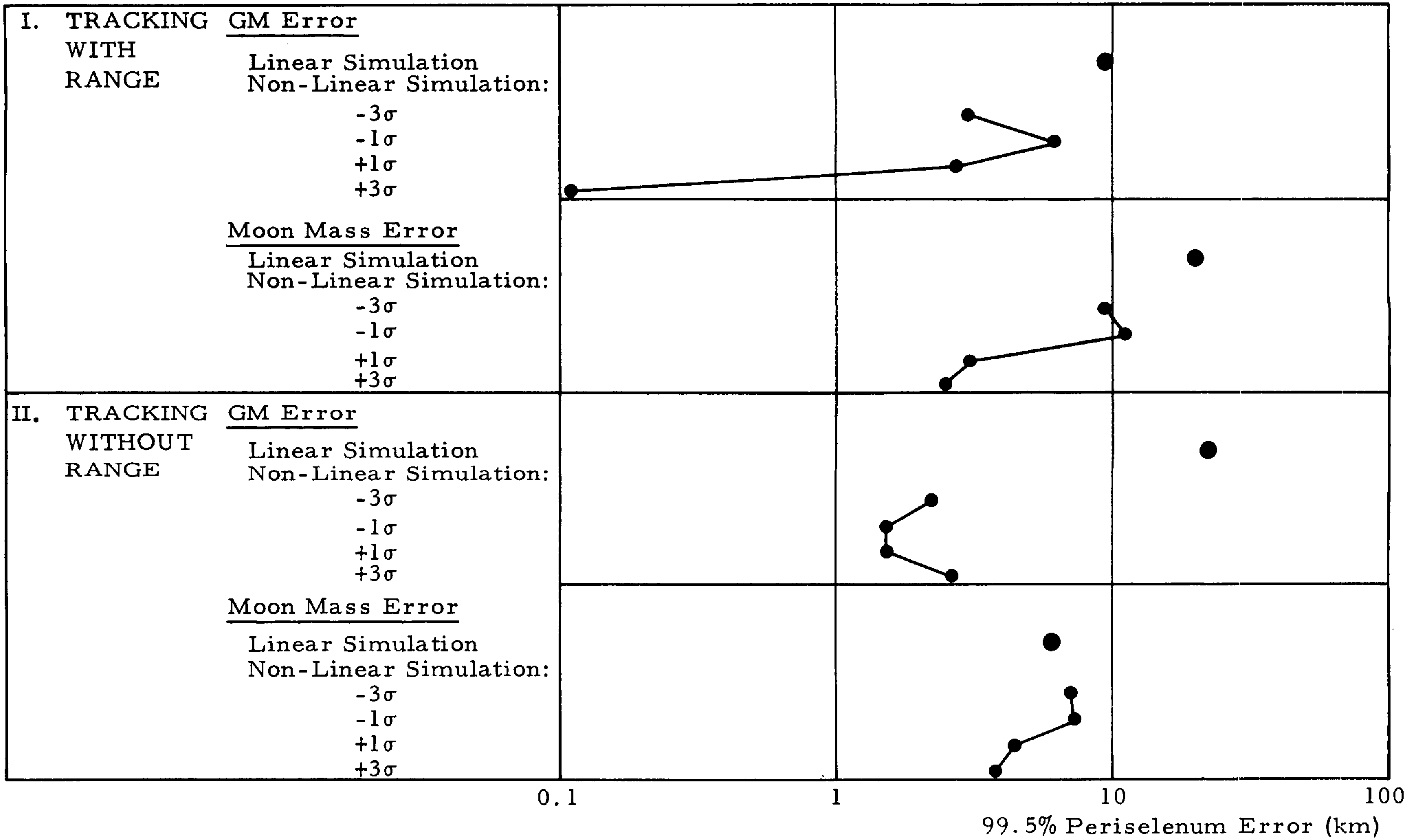


FIG. 4 LINEAR AND NON-LINEAR SIMULATION COMPARISON EARTH-MOON TRANSFER TO LUNAR ORBIT

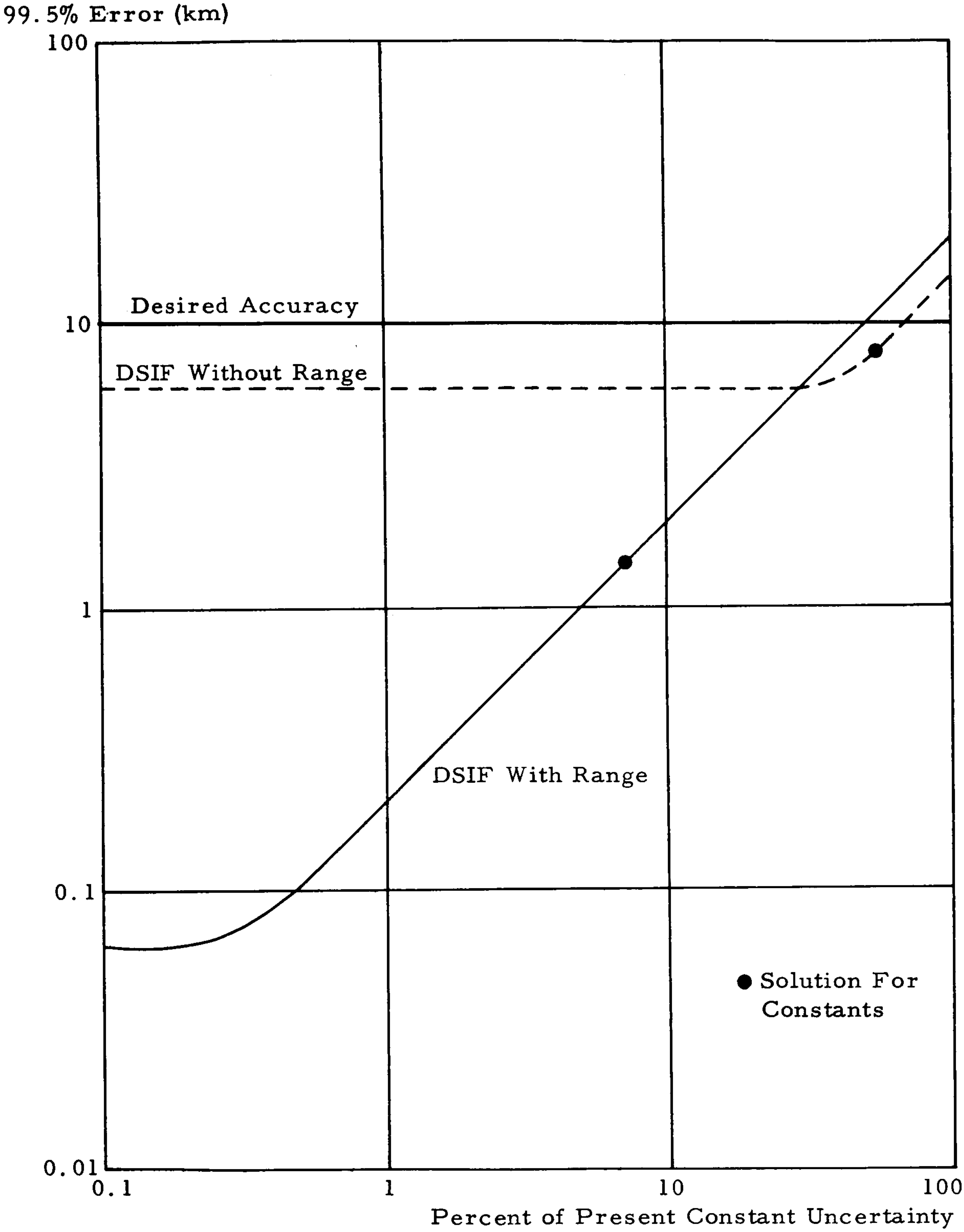


FIG. 5 LUNAR ORBIT MODE
PARKING ORBIT PERISELENUM ERROR

TABLE V

PERISELENUM ERROR CONTRIBUTIONS

	Error Sources	3σ Periselenum Error (km) With Range Tracking	3σ Periselenum Error (km) Without Range Tracking
I.	Random Observational	0.06	6
II.	Systematic		
	Station Coordinates	16	11
	Earth Gravitation Constant	6*	3*
	Moon Mass	10*	7*
	Velocity of Light	8	2
III.	Total Random Observational and Systematic Errors	21*	15*
IV.	Idealized Solution for Constants	2	8

* Adjusted to agree with non-linear simulation

on the error if this time restriction is relaxed and tracking is continued closer to the moon. This graph is based on the linear simulation, but the adjustment required for the systematic error level to yield agreement with the non-linear simulations is indicated. The accuracy level can be improved beyond that shown previously by about a factor of two if tracking data can be used up to about two hours before the braking maneuver. These results may be particularly altered by midcourse maneuvers (see Para. III F).

2. Landing From Lunar Parking Orbit. Following establishment of a lunar parking orbit, the sequence of events shown in Figure 7 has been assumed. The parking orbit ephemeris may be determined by tracking over a number of revolutions. On each revolution approximately 1.2 hours of earth DSIF tracking can be obtained (for a 185 km orbit), while the remaining 0.8 hours of each revolution is occulted by the moon. If a lunar surface beacon is available, approximately 10 minutes of tracking can be obtained on each revolution.

At the end of the final revolution in the parking orbit, a maneuver is performed to place the vehicle in a Hohmann ellipse with a periselenum of about 20 km. When the periselenum is reached about one hour after the maneuver, a final powered descent is begun to the surface.

The landing errors given will be due to tracking errors only, not including the error in performance of commanded maneuvers. The values given also do not show the influence of having guidance with a beacon during the terminal landing phase. Terminal guidance with a beacon would mean that landing errors on the order of 20 to 30 km from the orbit determination could be removed during the terminal descent with little performance penalty.

The analysis of tracking accuracy in the lunar orbit has been performed through linear simulation only. Computer program limitations precluded an exact non-linear simulation. However, the expected non-linearities are less severe than those encountered in analysis of earth-moon transfer tracking.

The landing errors will be given as the semimajor axis of the 99.5% confidence ellipse on the lunar surface, as used previously for the direct mode landing.

It has been assumed that the Hohmann maneuver will take place while the vehicle is occulted from the earth. The maneuver must be commanded before occultation takes place, and tracking for navigational purposes must cease some time earlier in order to allow time for a final orbit determination.

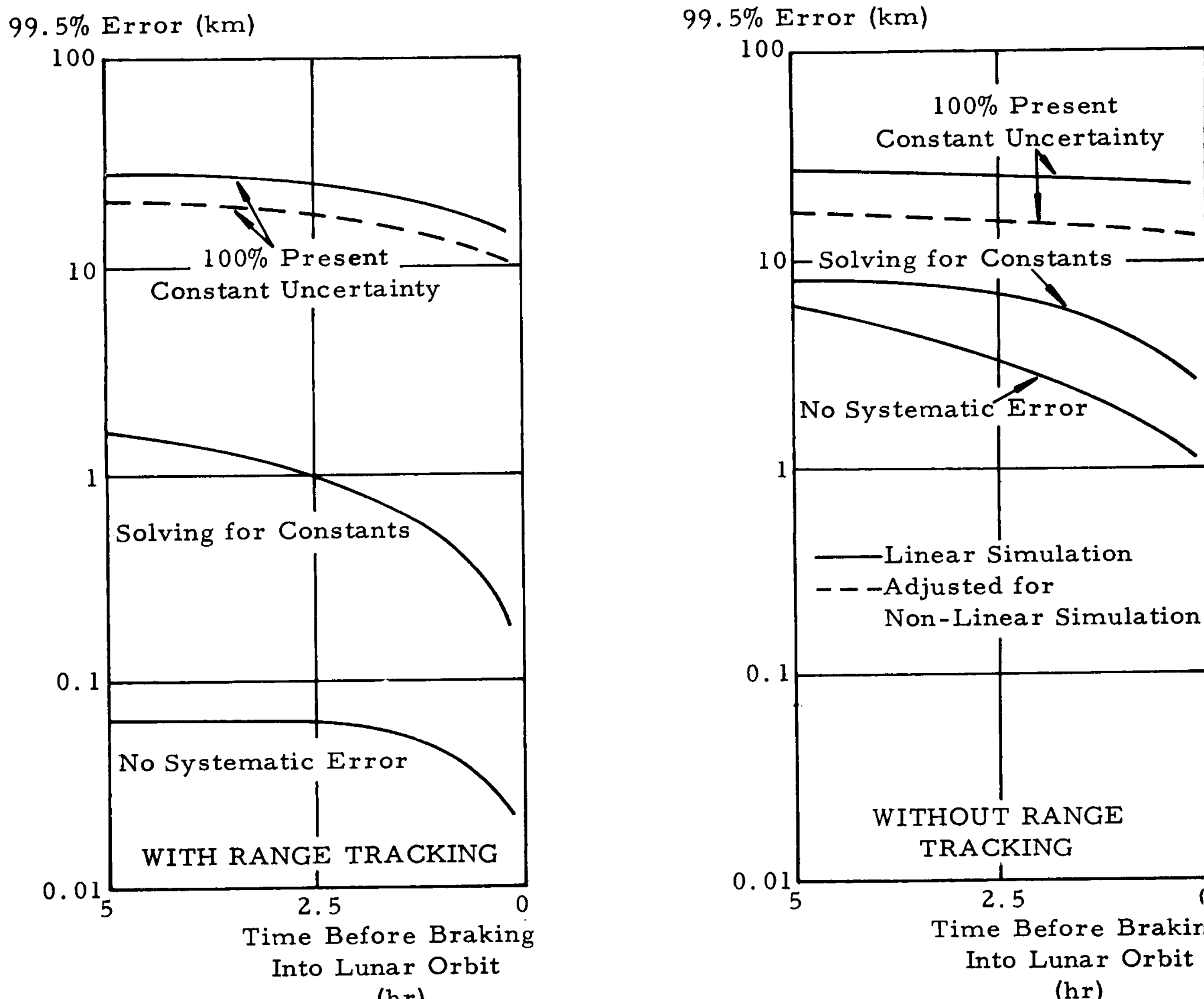
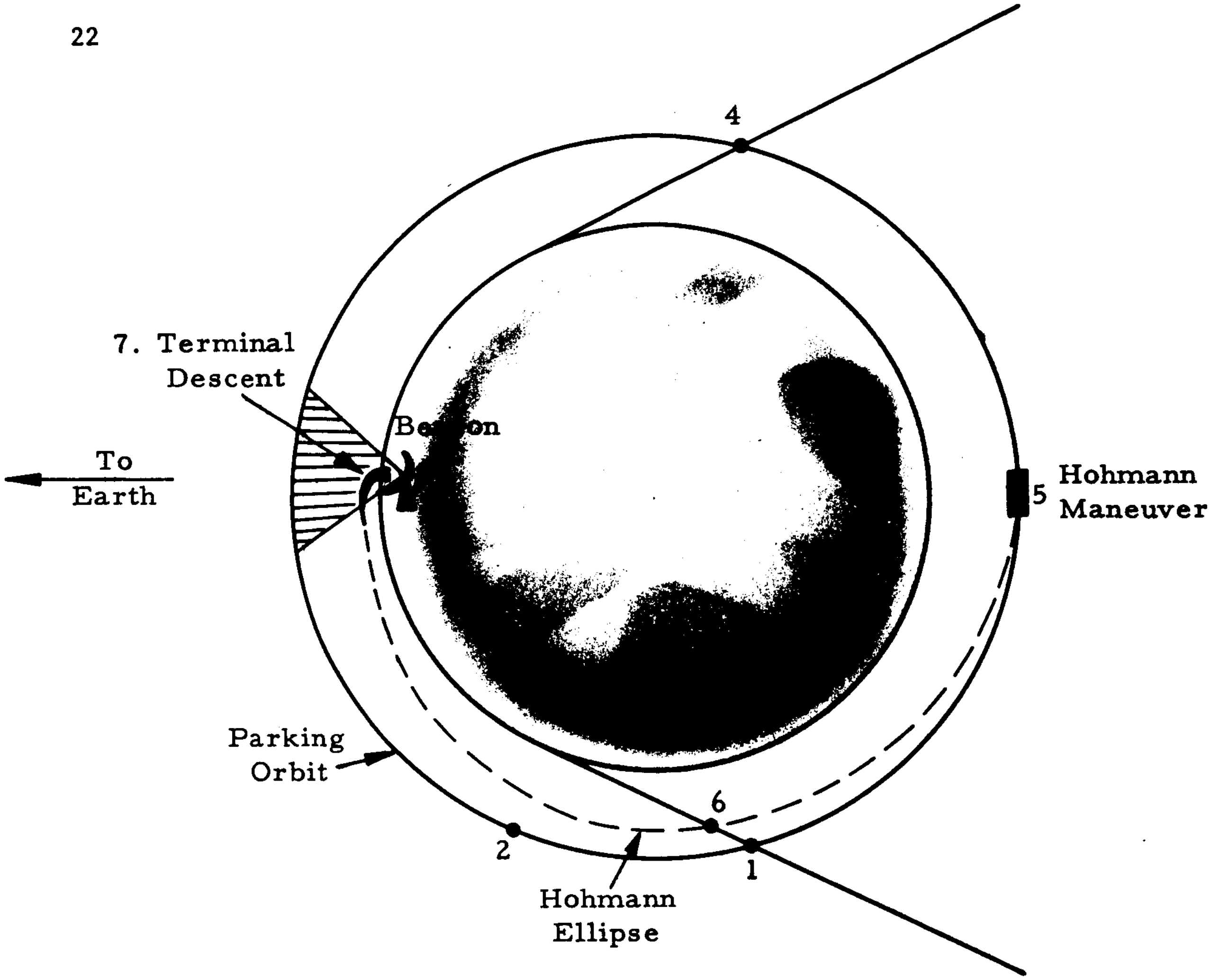


FIG. 6 PARKING ORBIT PERISELENUM ERROR TRACKING TIME EFFECT



Parking Orbit Tracking Sequence

- DSIF Tracking ~ 1.2 hr/orbit
  (Orbital Period ~ 2.0 hr)
- Lunar Beacon Tracking ~ 0.2 hr/orbit

Command Orbit Sequence

1. Start DSIF Tracking	(0 hr)
2. Cease Tracking	(0.2 hr)
3. Transmit Hohmann Command	(<1.2 hr)
4. Vehicle Occulation	(1.2 hr)
5. Hohmann Maneuver	(1.6 hr)
6. Vehicle Visible	(2.0 hr)
7. Terminal Descent	(2.6 hr)

FIG. 7 LUNAR ORBIT SCHEMATIC

In the results to be shown, one hour has been allowed for the final orbit determination and the Hohmann command transmission before the vehicle is occulted by the moon. This, in turn, means that only about 0.2 hours of DSIF tracking and no lunar beacon tracking is used during the final revolution in the parking orbit before the landing sequence is begun. Relaxation of the 1 hour requirement to 15 minutes would mean that the accuracy levels which will be shown could be achieved one revolution earlier than indicated. However, ample time for the final orbit determination command sequence must be provided, particularly if it is necessary to solve for some physical constants.

The landing accuracy which can be achieved through DSIF tracking only is shown in Figure 8 for the cases with and without range. It is assumed here that the prior information about the parking orbit obtained during the earth-moon transfer is utilized. The achievable landing error solving for constants will lie above the idealized solution level shown, due to the presence of other systematic errors not considered and the practical problems of solution. The desired accuracy level of about 2 km can be achieved, however, within two orbits provided ranging is used and knowledge of the physical constants is improved.

The importance of the prior orbit information gained during the earth-moon transfer is shown in Figure 9. The principal effect of not using this information when ranging is also not used is the requirement of two to three additional revolutions to reach a given accuracy level. However, after four to five revolutions, about the same accuracy level is reached. When ranging is used, the accuracy level after five revolutions can be reduced by an order of magnitude through the use of the transfer information. Two revolutions in the parking orbit are required to achieve the same level of accuracy provided only by transfer information.

The value of a lunar beacon is indicated in Figures 10 and 11. In Figure 10, ranging is assumed by both the DSIF and beacon. The one-revolution errors with and without beacon are identical because of the one-hour time restriction previously described, which means that no beacon data is used if a landing is performed after the first orbital revolution. Including the beacon pass would shift the results by one revolution. The beacon provides a significant increase in accuracy if an attempt is made to solve for physical constants, although the coordinates of the beacon are additional error sources and unknowns which are included in the results. However, the results shown assume no use of earth-moon transfer information. If this information is included, addition of the beacon data ceases to lower the error level significantly when ranging is used.

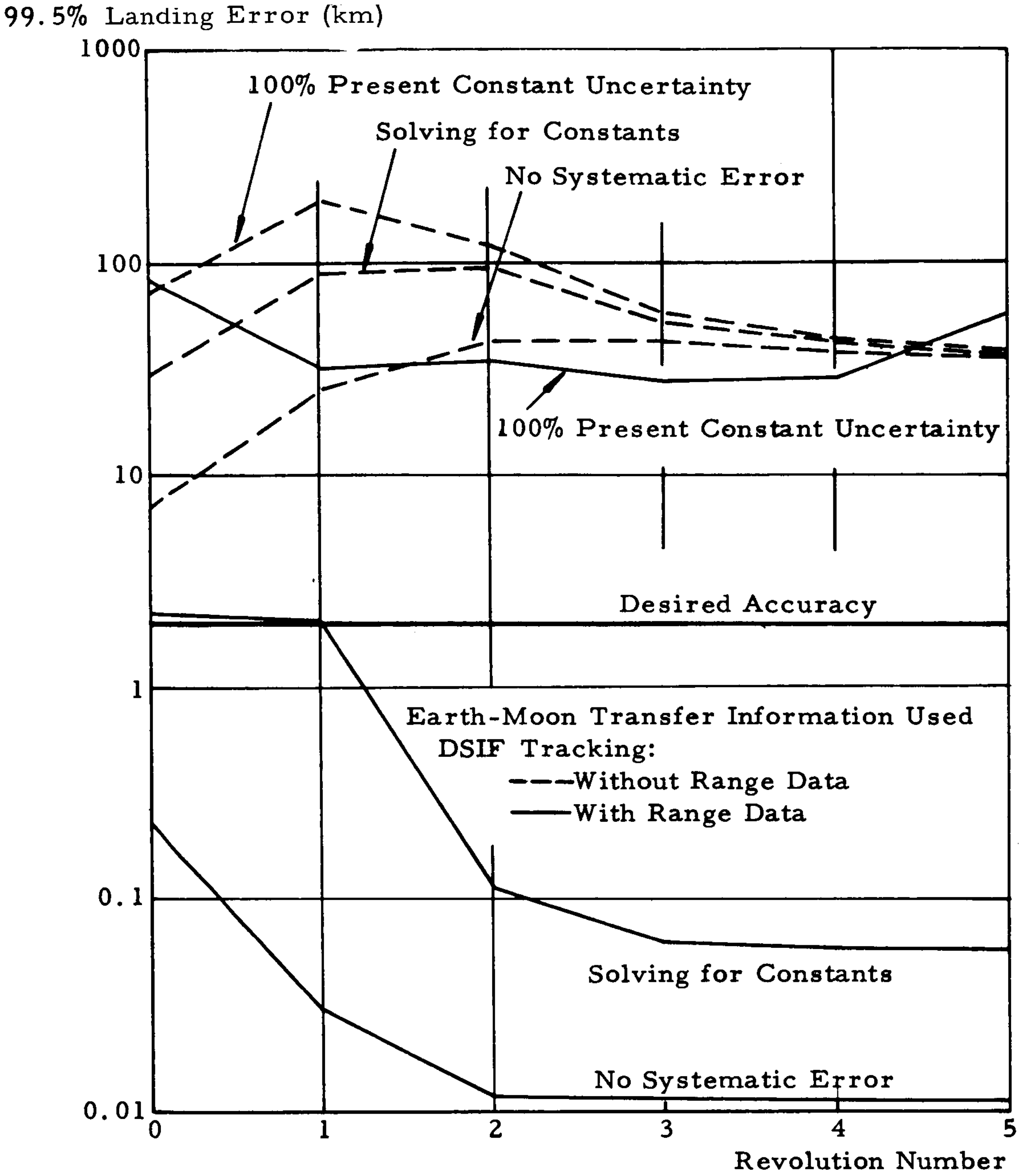


FIG. 8 TRACKING WITH AND WITHOUT RANGE DATA

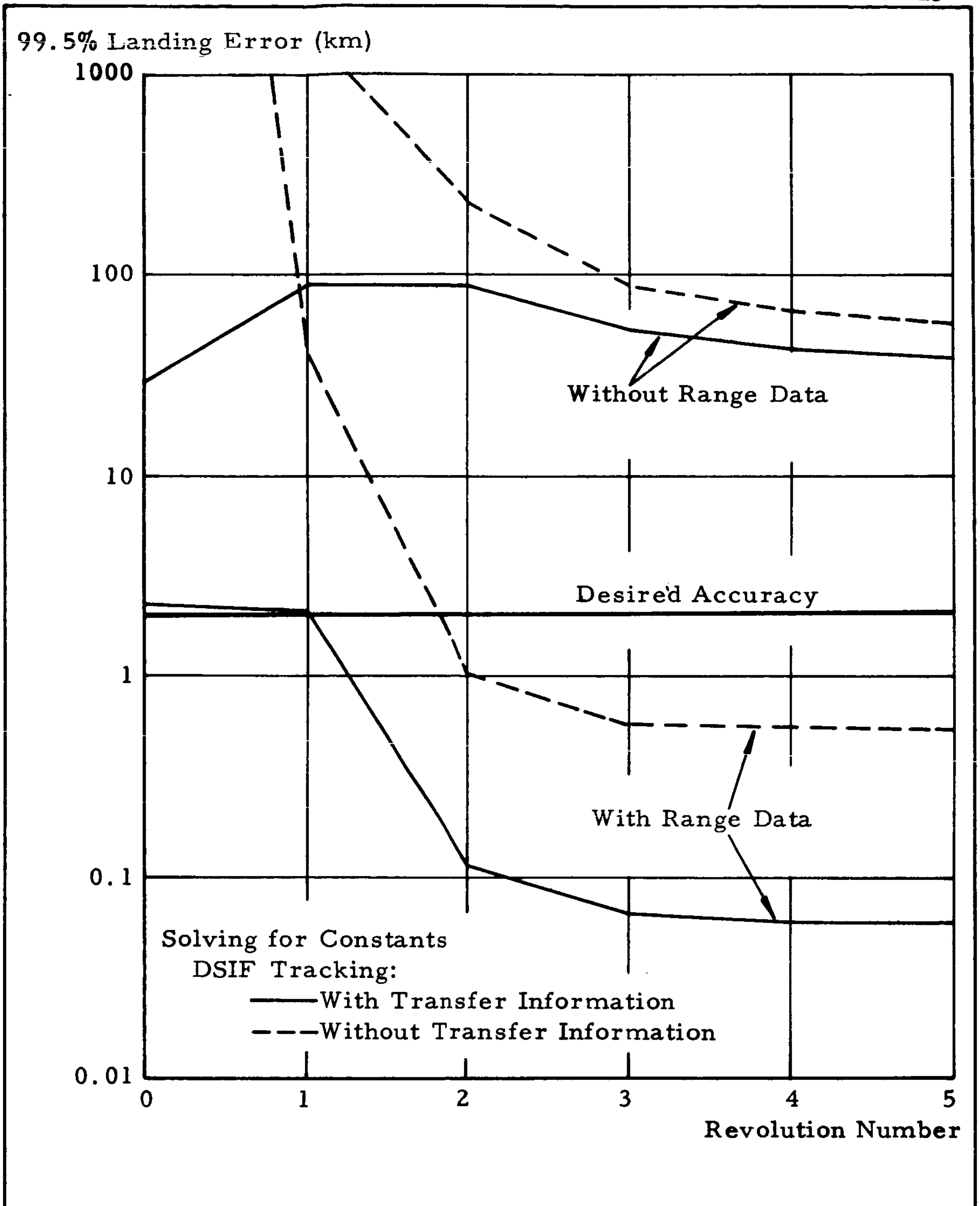


FIG.9 EFFECT OF EARTH-MOON TRANSFER INFORMATION

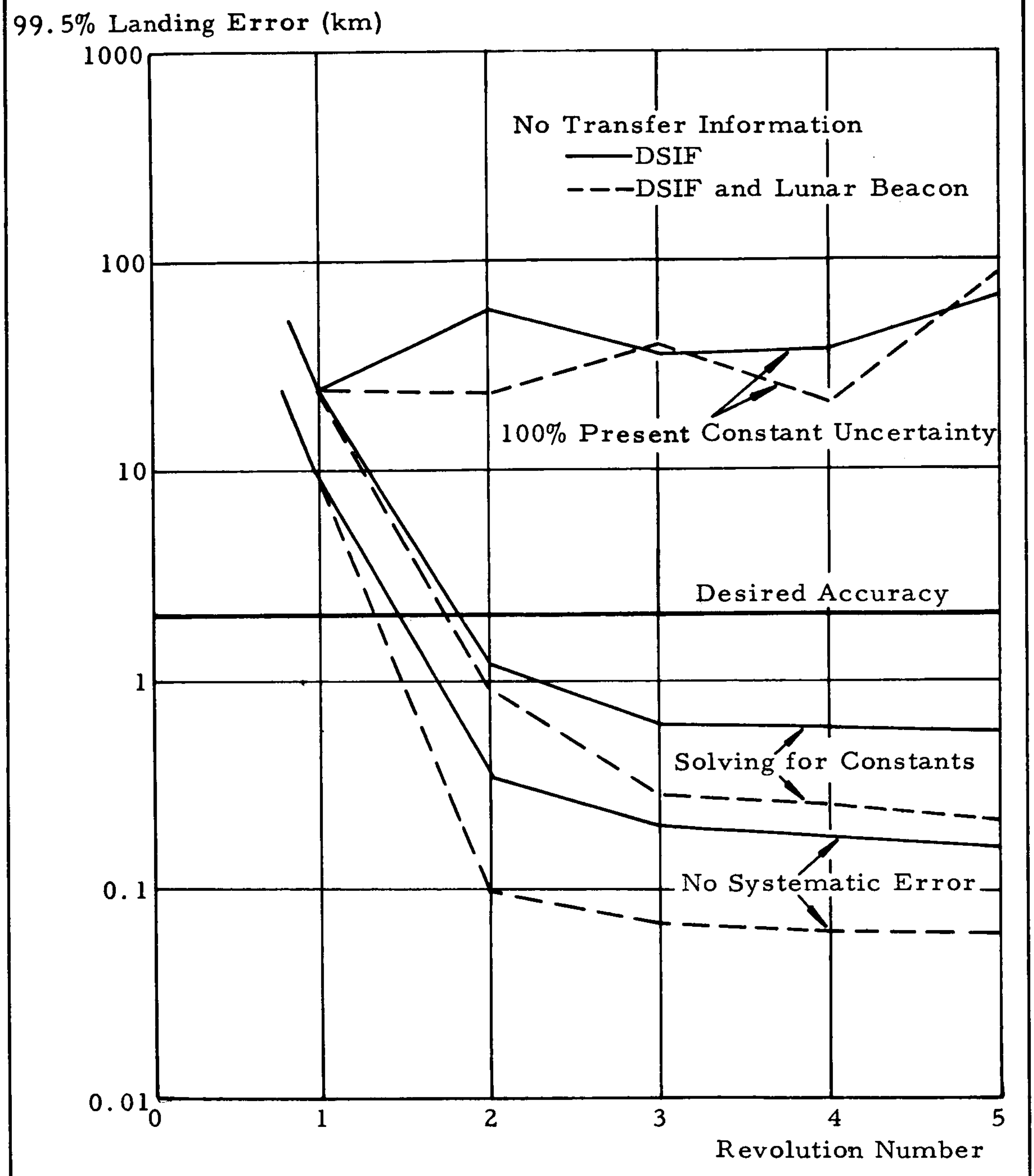


FIG. 10 EFFECT OF LUNAR BEACON WITH RANGE TRACKING

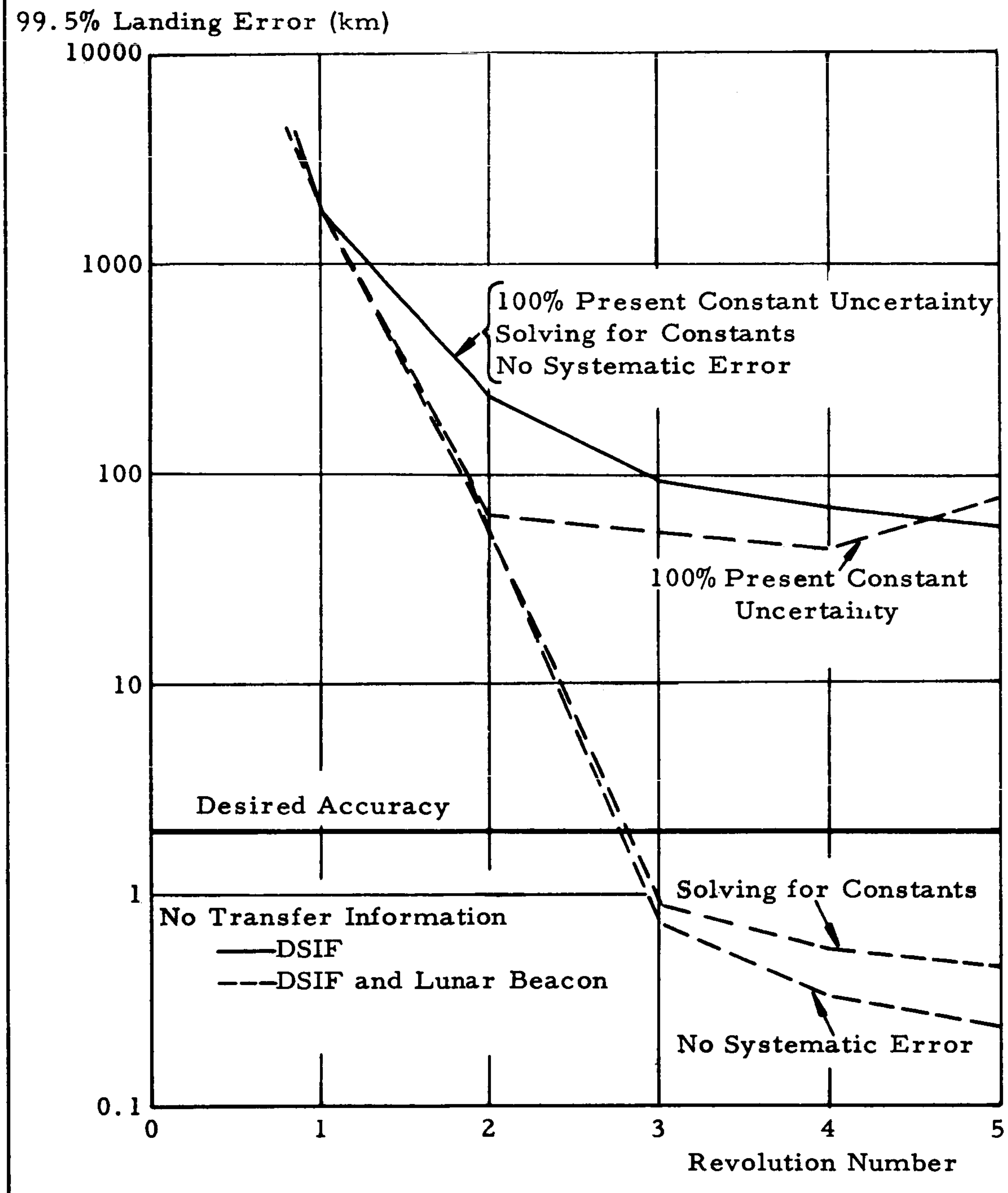


FIG.11 EFFECT OF LUNAR BEACON WITHOUT RANGE TRACKING

The case that ranging is not used for either the DSIF or beacon is seen in Figure 11. Note that the solid curve represents all three cases: considering random observational error, the present constant uncertainties, and solving for these constants. This occurs because the systematic error considered is less than that due to random observational error. Without ranging, the value of the beacon is much greater. The desired accuracy level is reached in the idealized solution for constants after three revolutions. The addition of earth-moon transfer information would further reduce the error levels shown.

Increase of the random observational errors assumed for the beacon by an order of magnitude reduces the accuracy gain obtained with the beacon to about half the amount shown.

It should be kept in mind that if a beacon is available for use as a navigational aid during the terminal powered descent to the surface, the desired landing accuracy level can be achieved with errors from orbit determination equivalent to about 20 to 30 km in the results shown here. In this case, the importance of accurate ranging data is reduced.

F. SUMMARY

The mission errors are summarized in Table VI. Systematic errors in the orbit determination are substantial, but planned instrumentation is sufficiently accurate to accomplish their reduction. However, additional error sources not considered in the analysis and practical problems of solution will limit this reduction. Achievable tracking accuracy will lie between the values derived for the present uncertainties of physical constants and those derived for an idealized simultaneous determination of constants and state variables. The effects of midcourse maneuvers on orbit determination will be indicated in Section III.

The desired landing accuracy of 2 km can be achieved in the direct mode if range data is available, and without range data if a lunar beacon is available during the terminal descent. Horizontal approach trajectories are less accurate than perpendicular approaches.

In the lunar orbit mode, the parking orbit can be established within permissible limits with or without range data, although ranging affords significantly greater accuracy.

Range data is required for the desired accuracy of landing from orbit if a lunar beacon is not available. With a beacon, however, ranging is not required. Due to the use of the beacon as a navigation aid during the terminal descent, the landing error incurred in the

TABLE VI

SUMMARY OF MISSION ERRORS
(99.5% Probability)

	Desired	Estimated With Range Data (km)		Estimated Without Range Data (km)	
		Solving for Constants	Present Constant Uncertainty	Solving for Constants	Present Constant Uncertainty
I. Direct Mode					
1. Perpendicular Approach Landing	2	0.3	18	4	7
2. Horizontal Approach Landing	2	1	84	11	28
II. Lunar Orbit Mode					
1. Periselenum of Parking Orbit	10	2	21	8	15
2. Landing (After 3 Parking Orbits)					
DSIF Only	2	0.1	30	50	60
DSIF and Lunar Beacon	20	0.1	40	0.8	60

orbit determination can be increased to 20 km. A minimum of two revolutions in the parking orbit are required with range data, and three without ranging. These requirements might be reduced by one revolution if the time required for the parking orbit determination and command sequence can be sufficiently minimized.

The landing errors shown in this table assume the information obtained during the earth-moon transit is used in the parking orbit determination. The landing error with present constant uncertainties is larger with the beacon than without due to the uncertainty in beacon location. This is true since the results shown refer to landing error referenced to the lunar surface, not the beacon.

For precise lunar landing operations, the many systematic error sources must be reduced. This can in most cases only be accomplished through flight experience, where the systematic errors must be separated from random observational errors. Ranging capability from earth to vehicle is necessary for this task, and is highly desirable for accomplishment of the logistic vehicle mission. Accumulated experience over several flights and improved techniques for real-time constant solution will be required to achieve the predicted system capability. Information obtained from flights without range measurement capability will be limited.

III. MIDCOURSE MANEUVER REQUIREMENTS

A. INTRODUCTION

Midcourse correction maneuvers during the earth-moon transit are required in order to accomplish the lunar logistics mission. A number of questions must be answered with regard to a midcourse correction scheme.

(a) What state variables must be controlled at lunar encounter to establish a satisfactory lunar parking orbit?

(b) How should these variables be controlled? Some may be controlled at lunar encounter through the braking maneuver, while others must be controlled during midcourse.

(c) What is the ΔV requirement for trajectory corrections applied during midcourse and during braking into lunar orbit?

(d) How many corrections should be applied and when should they be applied?

(e) What is the effect of the midcourse maneuvers upon the terminal accuracy of orbit determination, and how accurately can the orbit be established?

These questions are highly interrelated. The present study does not claim to answer these questions. However, certain tentative conclusions are indicated and directions for thorough analysis are indicated.

The present study has been limited to consideration of landing through a lunar parking orbit, so that the purpose of midcourse control is to establish the parking orbit. DSIF tracking with ranging capability has been assumed (see Par. IIB for details). Only one transit trajectory has been studied, the same used in the tracking analysis presented in Par. IIE. The analysis has been performed throughout with linear guidance equations and error theory, although Monte Carlo analysis is felt to be required for realism of results.

B. LUNAR ARRIVAL ERROR WITHOUT MIDCOURSE CORRECTIONS

Three categories of error sources cause the vehicle to deviate from the desired flight path and create a need for midcourse corrections.

(a) Injection Errors. Due to vehicle guidance and performance errors, the state variables at injection into the earth-moon transit may not be those desired. The 3σ error in the lunar close approach distance due to injection guidance error is about 510 km for the Saturn guidance system, assuming the vehicle remains one full revolution in an earth parking orbit without guidance updating. The full covariance matrix of injection errors assumed and their causes are described in Reference 1.

(b) Trajectory Prediction Error (Physical Constants). Due to uncertainty in physical constants such as the earth gravitation constant (GM), the actual vehicle flight path for known injection conditions may deviate from that predicted. The deviation, however, will be measured by tracking during the transit. If the physical constant contributing an error is included as an unknown in the orbit determination process, its true value will be determined; if it is not included, the flight path deviation will be attributed to erroneous injection conditions. In either case, however, a corrective maneuver must be performed to achieve the desired conditions at lunar encounter. The 3σ error in predicting the lunar close approach distance from earth injection conditions due to the present uncertainty in GM (See Table I) is 104 km, while that due to the moon mass uncertainty is only 1 km. Additional errors of this type may also occur due to incompletely known forces acting on the vehicle, such as impulses received from the vehicle altitude control system or gas leaks.

(c) Tracking Errors. The deviation of the actual flight path from that desired, caused by errors described in (a) and (b), is measured by tracking and orbit determination during the transit. Errors in the tracking and orbit determination process may create erroneous deviations which cannot be distinguished from the true deviations. Systematic tracking errors, such as earth station coordinate errors and the velocity of light, are important in this regard. These errors limit the accuracy with which the lunar orbit can be achieved and impose additional propellant requirements for the midcourse maneuvers. The magnitude of error in lunar close approach distance due to several such error sources is shown in Table VII.

C. STATE VARIABLES TO BE CONTROLLED

For the logistics vehicle mission, it is desirable to control the periselenum, apselenum, inclination, and nodal position of the lunar parking orbit. One feature of the orbit mode landing is that the accuracy of control of the earth-moon transit of parking orbit establishment does not directly affect the accuracy of lunar landing. However, for efficiency and reliability in the mission profile, the transit should be controlled as accurately as possible, reserving the flexibility of the lunar parking orbit for use in control of malfunctions and unanticipated errors.

TABLE VII

LUNAR CLOSE APPROACH DISTANCE ERROR

Error Source	Error Category (Par. IIIB)	3σ Approach Altitude Error (km)
Earth Injection Guidance Error	a	510
Earth Gravitation Constant	b	104
Moon Mass	b	1
Velocity of Light	c	3
Station Coordinates	c	9

The periselenum and apselenum are controlled to produce a near circular orbit (for simplification of the mission profile in orbit) at a desired altitude. The altitude is determined by performance consideration, safety in establishment, and convenience in the landing profile, including viewpoints such as the visibility of the landing site. For a nominal circular orbit of 200 km, variations in apselenum and periselenum heights of perhaps 20 km are immaterial, and larger deviations may be tolerated.

The inclination and nodal position of the orbit must be controlled to insure access to the landing site from the parking orbit. The relative importance of these two parameters depends upon whether the landing site is near the equator (nodal position more important) or near the maximum latitude covered by the orbit (inclination more important). The importance of both parameters depends upon the absolute inclination of the orbit, the maneuvering capability permitted during the descent from lunar orbit and the selected landing site. A variation of one degree in either parameter appears negligible.

Nominal condition equations for control of the lunar parking orbit might be expressed as

$$\begin{aligned} \delta A &= (A - A_o) = 0 \\ e &= 0 \\ \delta I &= (I - I_o) = 0 \\ \delta\lambda_N &= (\lambda_N - \lambda_{No}) = 0 \end{aligned} \qquad (1)$$

where A is the semi-major axis of the orbit, e is the eccentricity, I the inclination, and λ_N the nodal longitude. The subscript o indicates a desired value.

These conditions on the orbit reflect into the following conditions on the state variables at injection into the orbit (after braking into orbit).

$$\begin{aligned} &\epsilon_i = 0 \\ &\delta R_i = 0 \\ &\delta V_i = 0 \\ &\frac{\partial I}{\partial \alpha_i}\,\delta\alpha_i + \frac{\partial I}{\partial \psi_i}\,\delta\psi_i = 0 \\ &\frac{\partial \lambda_N}{\partial \alpha_i}\,\delta\alpha_i + \frac{\partial \lambda_N}{\partial \psi_i}\,\delta\psi_i + \frac{\partial \lambda_N}{\partial \lambda_i}\,\delta\lambda_i = 0 \end{aligned} \qquad (2)$$

where

R = altitude
V = velocity
α = velocity azimuth angle
ϵ = velocity elevation angle
ψ = latitude
λ = longitude

and the subscript i denotes values at injection. The four conditions (1) on the orbit reflect into five conditions (2) on the injection state variables. While the equalities in (1) and (2) need not be rigidly enforced (as previously noted), it is convenient to work with them, remembering that some relatively substantial errors can be permitted.

D. METHOD OF STATE VARIABLE CONTROL

1. Control Variables. In order to satisfy the five conditions (2) on the state variables at lunar orbital injection, there are a number of control variables which may be used. These are summarized in Table VIII. There may in general be more than two midcourse maneuvers. However, two are sufficient for the present discussion, and will be further justified later.

In Table VIII there are twelve control variables listed to satisfy the five conditions (2). The variables of the braking maneuver can control only the injection velocity vector, except that t_B may be used for control of altitude.

There are additional constraints on the twelve variables (e.g., on time of midcourse maneuvers), limiting their range of variation. Several schemes could still be developed, however, to satisfy the five condition equations by means of the twelve variables. An optimum could be obtained by combining the five condition equations with additional constraint equations upon the variables, and with condition equations to minimize the propellants required for midcourse and braking maneuvers. A more empirical approach has been followed at present in order to observe general characteristics. The relatively small midcourse propellant requirements that will be required for the logistics vehicle negates the need for extreme optimization. The scheme can in the long run be designed equally for operational convenience and reliability as for fuel economy.

A number of the variables in Table VIII can be practically eliminated. Since the second midcourse maneuver must eliminate errors in execution of the first maneuver, it cannot be regarded as an independent source of control. It will therefore be considered as a repetition of the first maneuver.

TABLE VIII

POSSIBLE CONTROL VARIABLES

Variable	Symbol
1. Braking into Lunar Orbit:	
Time of braking	t_B
Magnitude of velocity increment	V_B
Azimuth angle of velocity increment	α_B
Elevation angle of velocity increment	ϵ_B
2. Second Midcourse Maneuver	
Time of maneuver	t_{M2}
Magnitude of velocity increment	V_{M2}
Azimuth angle of velocity increment	α_{M2}
Elevation angle of velocity increment	ϵ_{M2}
3. First Midcourse Maneuver	
Time of maneuver	t_{M1}
Magnitude of velocity increment	V_{M1}
Azimuth angle of velocity increment	α_{M1}
Elevation angle of velocity increment	ϵ_{M1}

The time of the first maneuver is constrained by the economics of fuel consumption and orbit determination accuracy. Within the limited variation permitted, its altering effect is limited and it is practically eliminated as a control variable.

Some argument can be made for use of the time of braking into lunar orbit. Varying the time would affect the altitude, velocity magnitude, and velocity elevation angle when braking is begun, and consequently the inplane orbit conditions [first three equations of (2)]. In combination with V_B and ϵ_B, it would permit control of all three inplane conditions at braking into orbit, which would be desirable for accuracy purposes. However, it is most economical to perform the braking at the time of lunar close approach since this insures that $\epsilon_i = 0$ and eliminates any necessity to turn the vehicle velocity vector in the vertical plane during the braking maneuver. Braking at close approach destroys the variable ϵ_B for control purposes, since it requires that $\epsilon_B = 0$ for a circular orbit.

The injection altitudes obtained by varying t_B are limited by the close approach distance of the transit trajectory. In order to make full use of t_B, it would be necessary to plan the nominal braking maneuver time t_{BO} some time before close approach, which would permit reducing the braking altitude by a variation of t_B. The variation in altitude and velocity elevation angle as a function of time near lunar close approach for a typical case is shown in Figure 12. The penalty for purchase of an altitude reduction possibility is about 1 m/s per km, assuming an impulsive braking maneuver. This ΔV penalty is the added velocity increment required to turn the vehicle velocity vector in the vertical plane during braking, due to the non-zero initial elevation angle (Figure 12). If it is desired to have the capability of reducing the braking altitude 10 km by variation of t_B from nominal, the nominal braking impulse is 10 m/s greater than would be required if braking were performed at close approach.

Since a price must be paid in the nominal trajectory to permit full use of t_B as a control variable, the braking is assumed to occur at time of lunar close approach, the most economical time. Since the velocity vector is horizontal at lunar close approach, ϵ_B must be zero in order to satisfy the condition $\epsilon_i = 0$. However, t_B will still be available to increase the braking altitude above the close approach distance if this should be required after the midcourse maneuvers. The ΔV cost for purchase of a braking altitude increase is again about 1 m/s per km.

After the above considerations, five control variables (V_B, α_B, V_M, α_M, and ϵ_M) are assumed to remain for control of the four condition equations (2). The condition $\epsilon_i = 0$ has been satisfied by choice of t_B and ϵ_B.

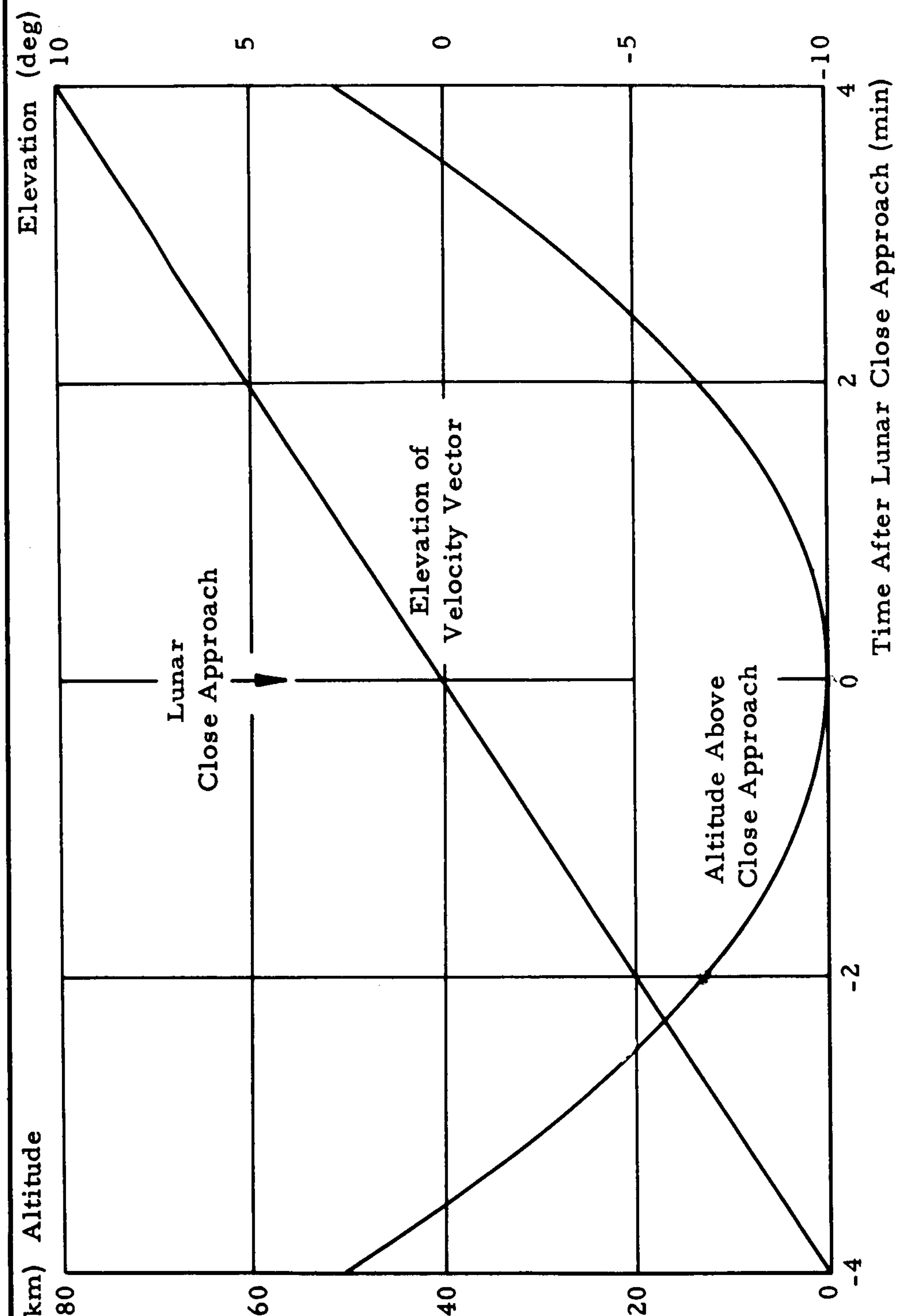


FIG.12 ALTITUDE AND ELEVATION ANGLE NEAR LUNAR CLOSE APPROACH

Since there now remains one more variable than condition equations, it is possible to impose another condition and solve the resulting conditions simultaneously for an optimum braking and midcourse maneuver. For simplicity in the present limited analysis, another approach was taken. The midcourse maneuver has been investigated independently of the braking maneuver.

2. Midcourse Control of Three Lunar Arrival Coordinates. One way to satisfy the condition equations (2) is to control R_i, ψ_i, and λ_i to their nominal values with midcourse corrections, leaving the arrival velocity vector to be corrected during the braking maneuver. This is equivalent to controlling the flight path to a completely nominal position and velocity at lunar orbit injection. This effectively imposes six control conditions rather than the five of (2). It was desired to determine the midcourse ΔV requirements for such a scheme and the effect of the midcourse correction on the uncontrolled velocity state variables at lunar arrival. The analysis was performed by the following technique using linear error theory.

Method of Analysis

The errors in lunar arrival state variables due to some source as, for example, injection guidance errors, are given by a covariance matrix Σ_G. The covariance matrix $\Sigma_{\Delta V}$ of three midcourse correction velocity components required to correct for the arrival errors Σ_G is given by

$$\Sigma_{\Delta V} = (\rho^{-1})^T \Sigma_{G1} (\rho^{-1}) \qquad (3)$$

where ρ is the three dimensional matrix of partial derivatives of the lunar arrival state variables to be controlled with respect to three velocity vector component changes at the time of midcourse correction. The matrix ρ , and hence the matrix $\Sigma_{\Delta V}$ is a function of time of midcourse correction. The matrix Σ_{G1} is the three dimensional covariance matrix of the state variables to be corrected, and is a submatrix of the complete state variable covariance matrix Σ_G.

The covariance matrix Σ_G' for state variable errors remaining after an assumed midcourse correction is computed as:

$$\Sigma_G' = \Sigma_G - B - B^T + \rho_T^T \Sigma_{\Delta V} \rho_T \qquad (4)$$

where ρ_T is the matrix of partial derivatives of all lunar arrival state variables with respect to the three velocity components of the midcourse correction.

The matrix B is

$$B = \Sigma_{G2}^T \rho^{-1} \rho_T$$

where Σ_{G2} is the submatrix of Σ_G containing the variances of the uncorrected state variables plus all covariances among the uncorrected state variables and between the corrected and uncorrected state variables.

All elements of Σ_G relating to the errors in the corrected state variables are identically zero. The non-zero elements of Σ_G' describe the errors in the uncorrected state variables due to the initial (guidance) errors plus the effect of the midcourse maneuver.

Results

The analysis was applied to characteristic injection errors of the Saturn guidance system after one full revolution in an earth parking orbit, as described in Reference 1. The state variable errors at lunar close approach due to these guidance errors are shown in Table IX. The 99.5% ΔV requirement to correct the three arrival coordinates as a function of the time in midcourse when the maneuver was assumed to be applied is shown in Figure 13. The 3σ errors in lunar arrival velocity magnitude V_i and velocity azimuth angle α_i after the maneuver are shown in Figure 14. No error is shown for ϵ_i since the errors are always referred to lunar close approach.

Two characteristics of Figures 13 and 14 are of interest. The ΔV requirement for a maneuver applied 20 hr after injection is quite large. Examination of the partial derivatives of the midcourse correction components with respect to the state variables being controlled (the matrix ρ^{-1}) revealed the reason. The direction of impulse application for a correction of an error in R_i was almost parallel to the direction for correction of λ_i. This singular situation requires very large impulses for simultaneous correction of both variables.

The characteristic of interest in Figure 14 is that the midcourse correction of R_i, ψ_i, and λ_i significantly also reduced the probable error in velocity and azimuth angle, except at the singular time of 20 hr. However, the error in arrival velocity is still quite large, requiring a possible correction ΔV at braking of 75 m/s if a single midcourse correction were performed at 10 hr.

3. Midcourse Control of Two Lunar Arrival Coordinates With Minimization of Propellant. The results shown in Par. III.D.2 led next to examination of a control scheme wherein two lunar arrival position coordinates, R_i and ψ_i, were corrected during midcourse. The third degree of freedom in the midcourse maneuver was used to satisfy a third condition of minimizing the propellant required for the correction. The analysis was again performed by an appropriate modification of the technique described in Par. III.D.2.

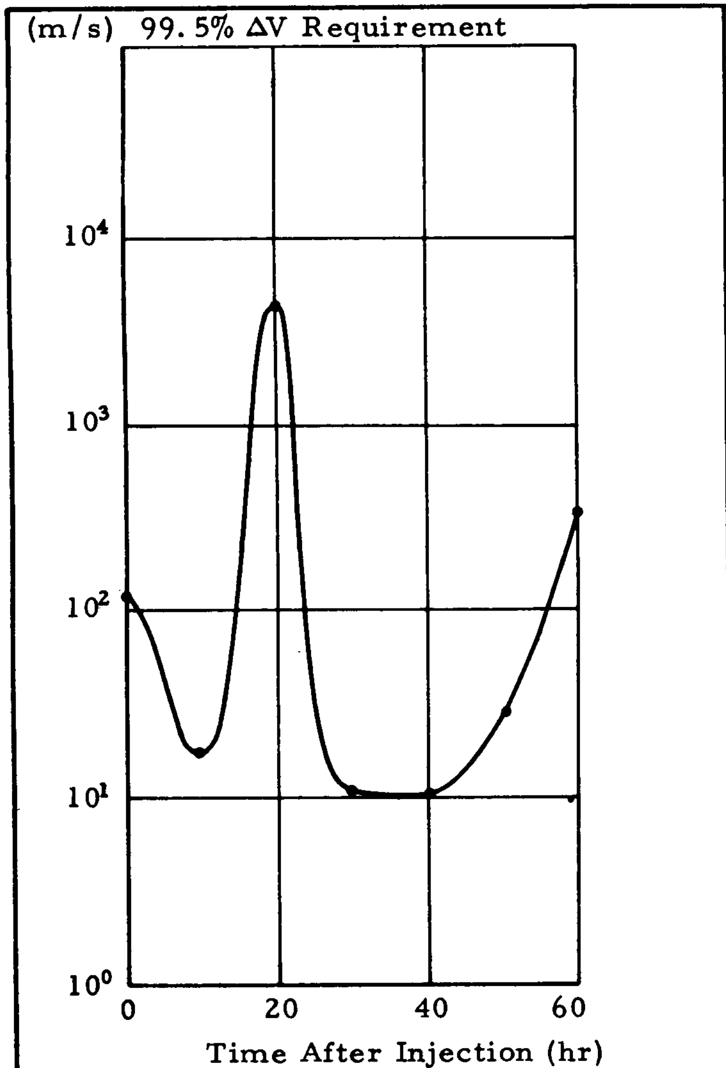


FIG.13 MIDCOURSE ΔV REQUIREMENT: CORRECTING R, ψ, λ

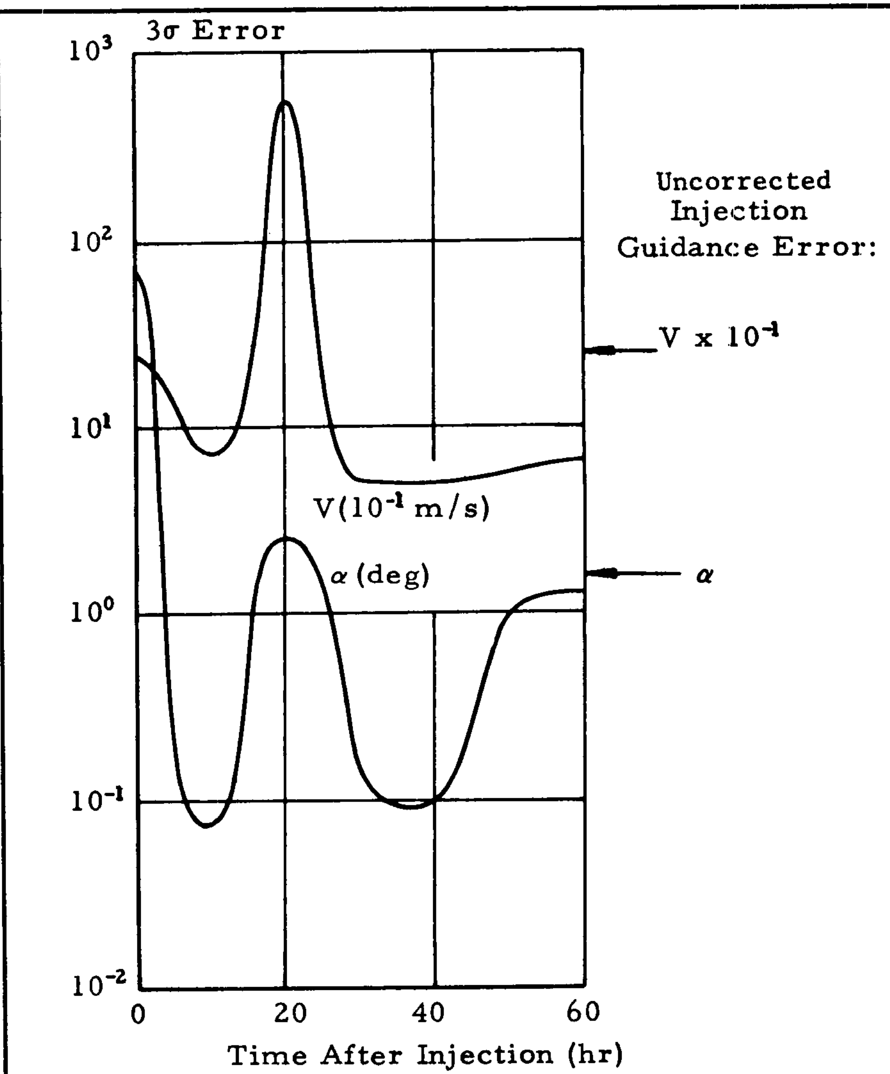


FIG.14 ERROR REMAINING IN V, α: AFTER MIDCOURSE CORRECTION OF R, Ψ, λ

The 99.5% ΔV requirement obtained for this scheme is shown in Figure 15, while the 3σ errors in the uncorrected variables λ_i, V_i, and α_i are shown in Figure 16. The ΔV requirement is significantly less than for correction of R_i, ψ_i, and λ_i. Substantial reductions are also produced in the errors in V_i, α_i, and λ_i. However, the error in V_i is still substantial, and requires a possible correction ΔV of 42 m/s at braking for a midcourse correction performed at 10 hr.

4. Midcourse Control of Lunar Arrival Altitude and Velocity Vector. A third scheme was investigated, assuming midcourse control of lunar close approach altitude R_i, velocity V_i, and velocity azimuth angle α_i. The method of analysis was as in Par. II.D.2. The midcourse ΔV requirement is shown in Figure 17, and the error in ψ_i and λ_i after the maneuver in Figure 18.

The ΔV requirement for this scheme increases steeply with time of maneuver, in contrast to the previous cases where a flat plateau was observed in the requirement during a considerable portion of flight. However, the correction ΔV required during the braking maneuver is greatly reduced. For a maneuver between 10 and 20 hr, the error in ψ_i and λ_i after the maneuver is quite small.

5. Summary of Control Schemes. The investigations thus far performed show only the trend of results. Definite conclusions should not be drawn until a variety of trajectories, possible error distributions to be corrected, and schemes of control are investigated. From the results thus far, it appears that the total corrective ΔV requirements during midcourse and braking maneuvers can be substantially reduced by proper choice of a midcourse correction scheme, in comparison to the requirement obtained if the scheme attempts to insure completely nominal lunar orbit injection conditions as in III.D.2. No significant degradation in accuracy of lunar orbit establishment would result. A simultaneous optimization of midcourse and braking maneuver corrections should be studied. This must take into account the higher specific impulse (and hence more economical fuel consumption) of the cryogenic propellants used in braking, as opposed to earth storable propellants used during midcourse. However, the total magnitude of corrective ΔV required is small enough not to require extreme concern with optimization for fuel consumption.

E. GENERAL CRITERIA FOR APPLICATION OF MIDCOURSE MANEUVERS

Criteria must be established for the decision of when to perform a midcourse maneuver. A maneuver should not be performed unless it is certain that the vehicle is deviating significantly from its desired flight path. Since the actual flight path is determined with a varying accuracy (generally improving with increasing flight time), this means

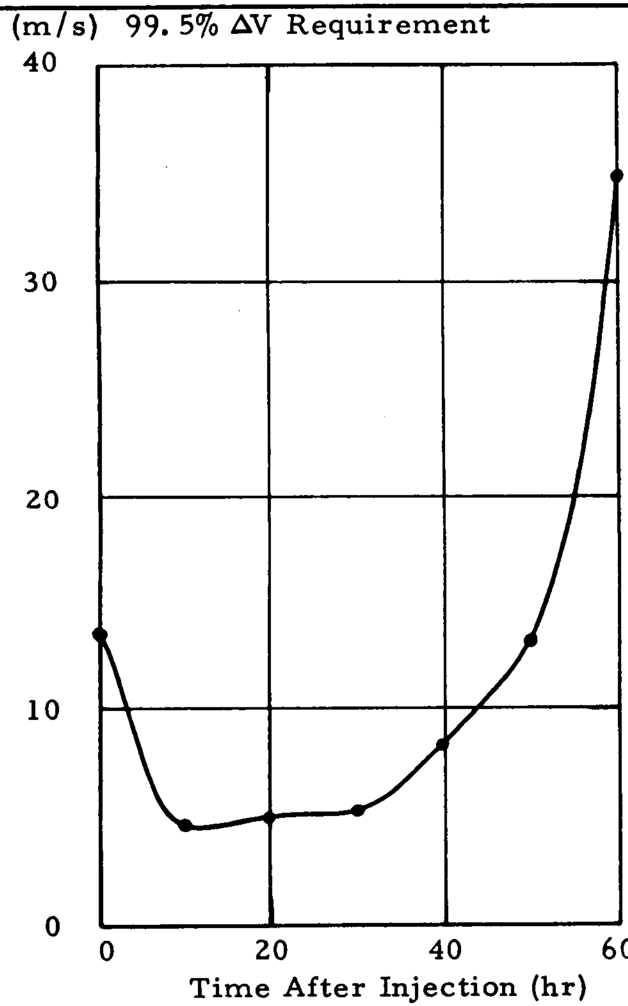


FIG.15 MIDCOURSE ΔV REQUIREMENT: CORRECTING R, ψ

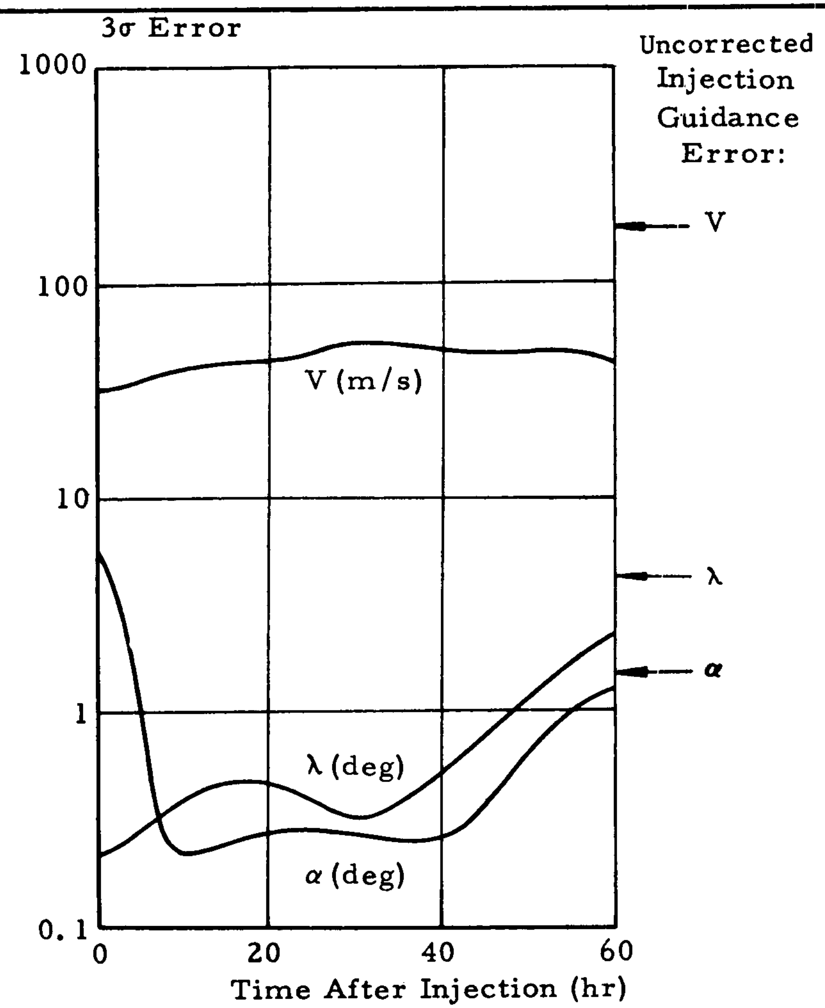


FIG.16 ERROR REMAINING IN V, λ, α: AFTER MIDCOURSE CORRECTION

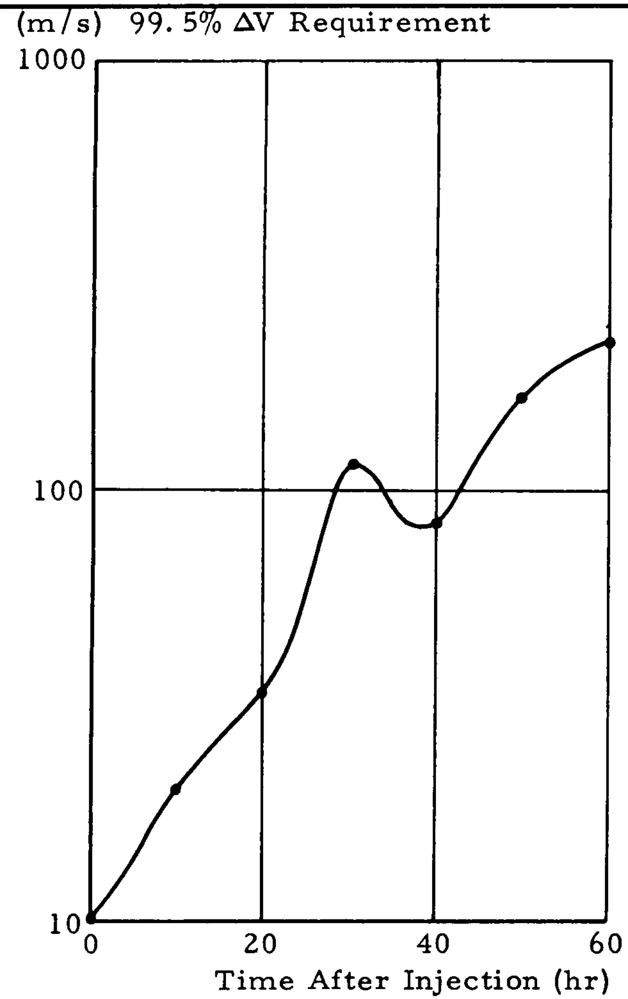


FIG.17 MIDCOURSE ΔV REQUIREMENT: CORRECTING R, V, α

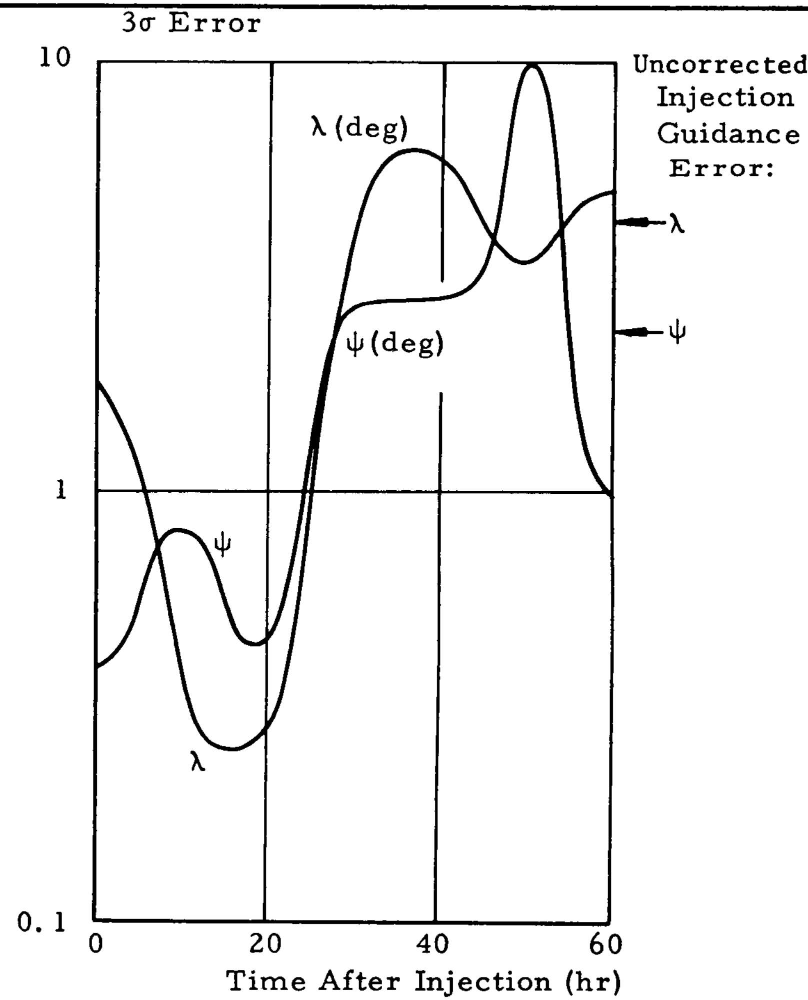


FIG.18 ERROR REMAINING IN ψ, λ: AFTER MIDCOURSE CORRECTION OF R, V, α

that the maneuver should not be performed until a time when the measured flight path deviation is a factor larger than the possible error in the measurement.

On the other hand, two considerations create a desire to perform the maneuver early. The later the maneuver is performed, in general the more expensive it is in fuel, although there is a reasonably broad period within which the nominal maneuver can be performed economically. Since the maneuver itself can only be performed and measured with a certain error, performance of the maneuver also degrades the accuracy with which the subsequent flight path can be predicted. In order to perform a subsequent midcourse maneuver and finally braking into lunar orbit, time must be allowed after each maneuver for tracking and redetermination of the flight path. For highest accuracy, a maximum of time must be allowed after the maneuver.

The proper time to perform a maneuver then depends upon a number of factors:

(a) The measured flight path deviation;

(b) The accuracy of flight path measurement;

(c) The amount of propellant available;

(d) The propellant cost of correcting the measured flight path deviation as a function of time;

(e) The number of midcourse maneuvers possible;

(f) The requirement to accurately perform a terminal braking maneuver;

(g) The accuracy with which a maneuver can be performed and measured.

Additional constraints may also arise, as from visibility requirements to a specific ground command station. An attempt to formulate decision tables for optimum control of maneuver time has been begun in Reference 2. Some tentative principal results indicated by this report, though derived from a crude and incomplete model of the process, are in general agreement with empirical results obtained in other lunar studies from flight simulations with parameter variations (e.g., Ref.3). These results are:

(a) Two maneuvers appear sufficient for most lunar missions;

(b) The first maneuver should occur about 10 hr after injection into the earth-moon transfer, the second in the time region of 30-50 hr;

(c) The measured flight deviation should be 3 to 10 times larger than its uncertainty at maneuver performance;

(d) Although the precise optimum maneuver control can only be determined for a specific case of flight deviations and other parameters, a properly chosen but fixed schedule can be followed without serious degradation of performance.

F. TENTATIVE MIDCOURSE PROFILE FOR THE LUNAR LOGISTICS VEHICLE

Although the technique presented in Reference 2 for formulation of optimum decision criteria should be pursued for definition of a final scheme, application of the tentative conclusions described in Par. III.E. leads to an instructive typical midcourse profile for the Lunar Logistics Vehicle. A Monte Carlo simulation of the midcourse profile under varying conditions would offer the most thorough results. The analysis given here, while considerably more crude, yields reasonable first indications of midcourse requirements.

It is assumed that two maneuvers will be performed, and the results under this assumption will be found completely satisfactory. Two maneuvers are required for most accurate control of the lunar approach, but more than two would create a problem in accurate redetermination of the flight path before the braking maneuver into lunar orbit.

The following discussion will consider only random errors in the tracking and orbit determination process. The effect of systematic errors will be considered later in Par. III.G.

The altitude, velocity, and azimuth angle at lunar close approach are assumed to be controlled by the midcourse as discussed in Par. III.D.4. However, the distance of close approach R_i will be used as a critical parameter for selection of maneuver times.

The magnitude of the 3σ error in R_i due to typical Saturn injection guidance error is 510 km, as shown previously in Table IX. Based on the injection guidance errors and the midcourse correction scheme selected, the first correction should occur as early as possible for minimization of propellant. The estimated 3σ error in orbit determination assuming DSIF tracking with ranging, and no systematic error sources, is shown in Figure 19. By 10 hr, the error in orbit determination and prediction of R_i is about 0.1 km, far less than the probable injection error. The first midcourse maneuver is assumed to be performed at 10 hr. The 3σ magnitude of the maneuver at this time is about 20 m/s.

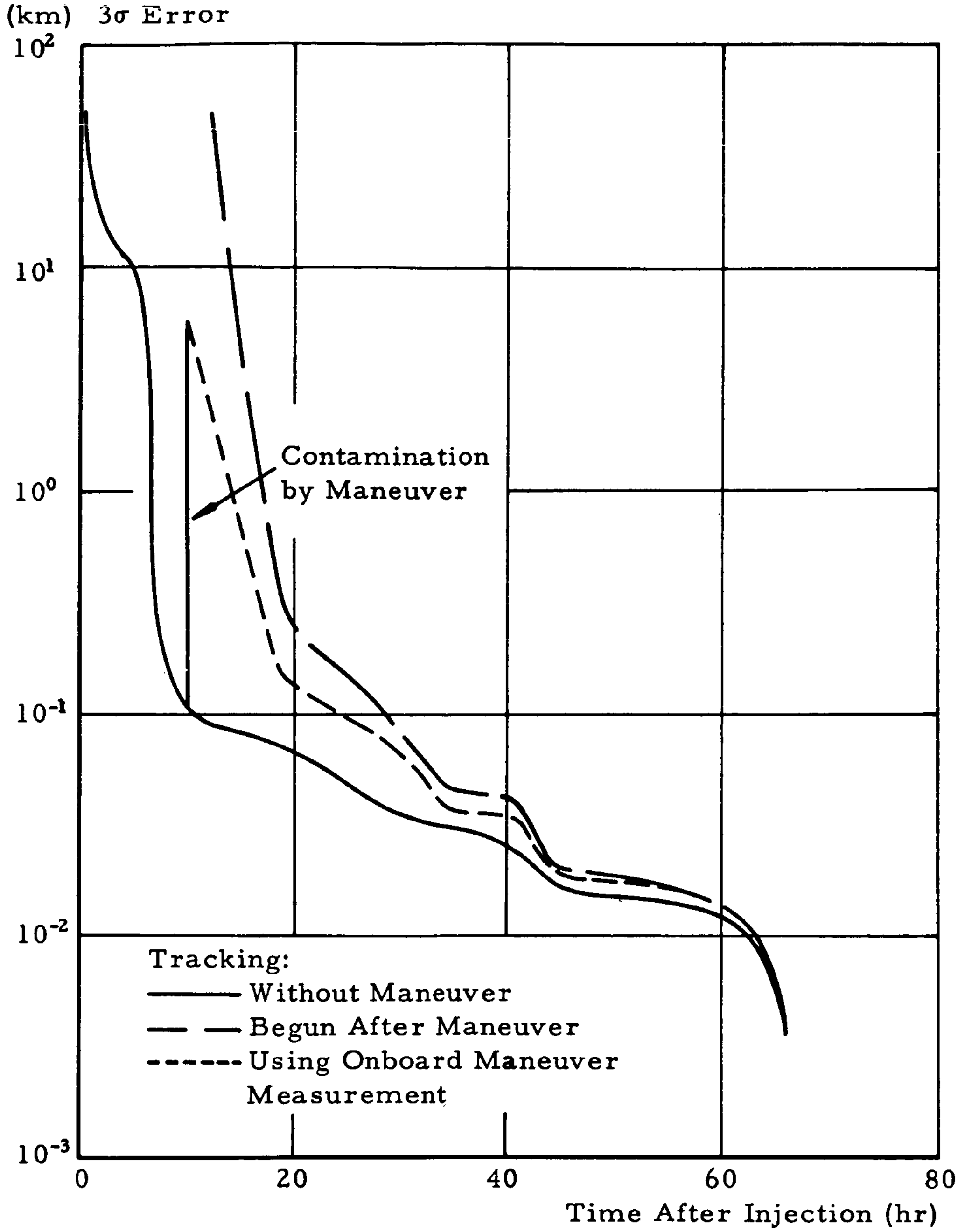


FIG.19 CLOSE APPROACH DISTANCE ERROR

TABLE IX

LUNAR CLOSE APPROACH ERRORS DUE TO EARTH INJECTION GUIDANCE ERRORS

3σ Errors:

Velocity Magnitude (V)	210 m/s
Azimuth of Velocity Vector (α)	1.7 deg
Altitude (R)	510 km
Geocentric Latitude (ψ)	2.3 deg
Longitude (λ)	4.2 deg

Error Correlation Coefficients:

$V\alpha$ = -0.2034
$V\epsilon$ = -0.8872
VR = -0.9888
$V\psi$ = -0.7142
$V\lambda$ = -0.9898
$\alpha\epsilon$ = -0.2140
αR = 0.2471
$\alpha\psi$ = -0.5331
$\alpha\lambda$ = 0.2963
ϵR = 0.8502
$\epsilon\psi$ = 0.9053
$\epsilon\lambda$ = 0.8313
$R\psi$ = 0.6877
$R\lambda$ = 0.9980
$\psi\lambda$ = 0.6501

The accuracy of orbit determination during the first hours of flight is sensitive to how soon after injection tracking data is obtained, due to the importance of data during the early period of rapid dynamic change. Variations performed in the injection location on the earth surface indicated errors ranging from about twice as large as shown at 10 hr to an order of magnitude smaller.

The maneuver will be measured by the onboard inertial guidance system. The magnitude of the maneuver should be measured to about 0.01%, while the direction is known within 0.2 deg. This uncertainty will contaminate the knowledge of the vehicle orbit, reducing the accuracy with which the lunar close approach can be predicted immediately after the maneuver to about 2 km (neglecting systematic errors in orbit determination). This is indicated by the vertical jump in Figure 19 at 10 hr. Assuming that the maneuver can be performed with the same accuracy as measured by the onboard guidance system (which is not completely correct), this also represents the remaining error in close approach distance which must be corrected by the second maneuver.

Following the first maneuver, the vehicle is again tracked and its flight path redetermined. About 14 hr are required to recover the same accuracy of orbit determination as before the maneuver. In order to recover this accuracy from ground tracking alone (independent of the onboard maneuver measurement) an additional 4 hr (total of 18 hr) after the maneuver would be required, if tracking is began after the maneuver. Some additional information may be gained from range rate information obtained during maneuver performance, and might reduce this lost time. Sufficient accuracy of orbit determination is reached in the time period of 30 to 50 hr after earth injection to permit performance of the second maneuver. The accuracy of knowledge of close approach distance at 50 hr is about 3 times better than at 30 hr, but the determination error is sufficiently small at 30 hr in relation to the possible flight path deviation to be corrected and in relation to the required precision of orbit establishment.

The 3σ magnitude of the second maneuver is shown in Figure 20 as a function of time of performance of the second maneuver. The propellant requirement increases by a factor of 1.6 between 30 and 50 hr, but the absolute magnitude is quite small, only 4 m/s at 50 hr.

Other considerations in choosing the time of the second maneuver must be considered also. The accuracy of orbit determination after the second maneuver is shown in Figure 21 for three cases, assuming the second maneuver is performed at 30, 40, or 50 hr.

The error in knowledge of the close approach altitude immediately after the maneuver (assumed to represent also the error in control of close approach distance) is reduced by a factor of about two

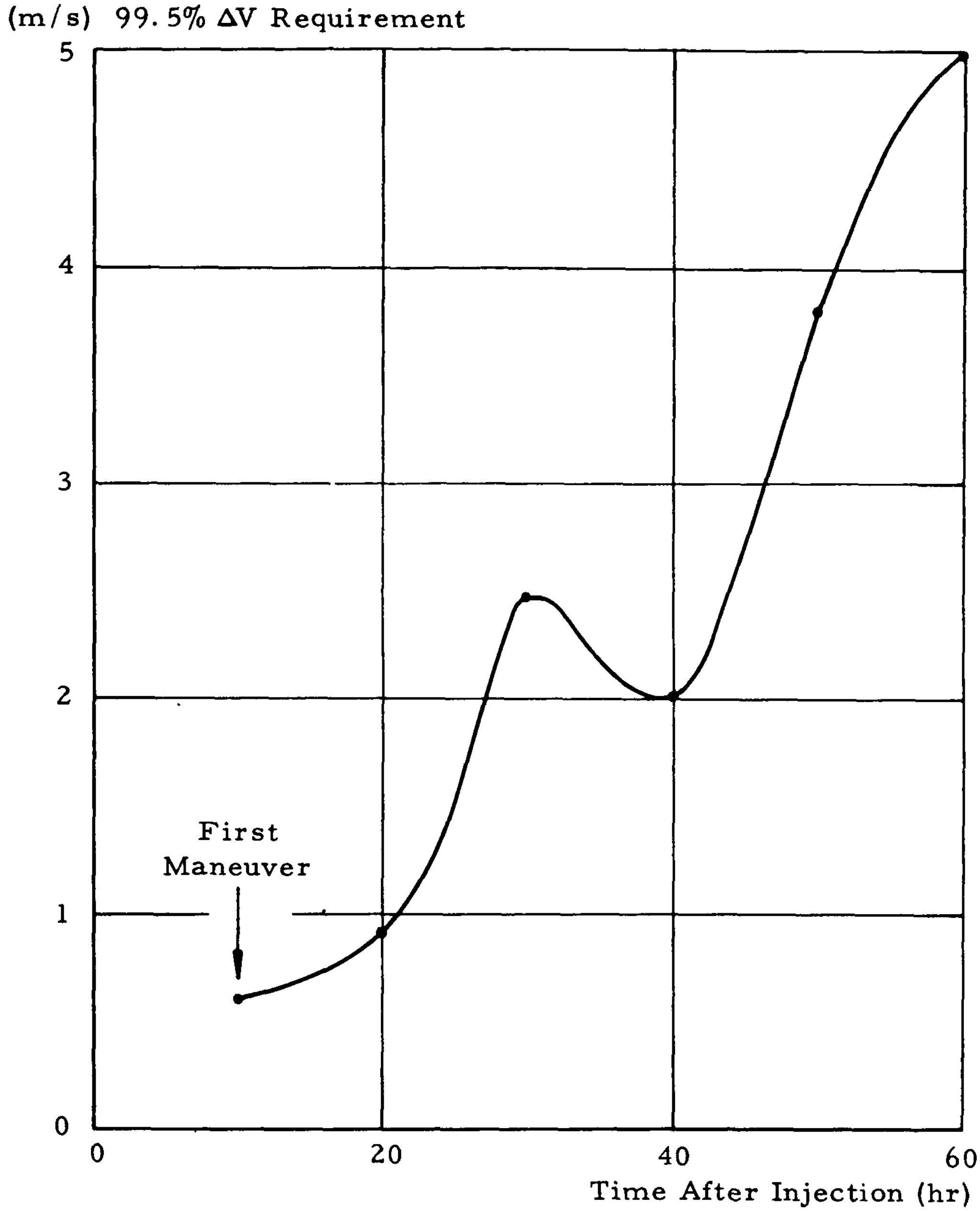


FIG.20 SECOND MIDCOURSE ΔV REQUIREMENT: CORRECTING R, V, α

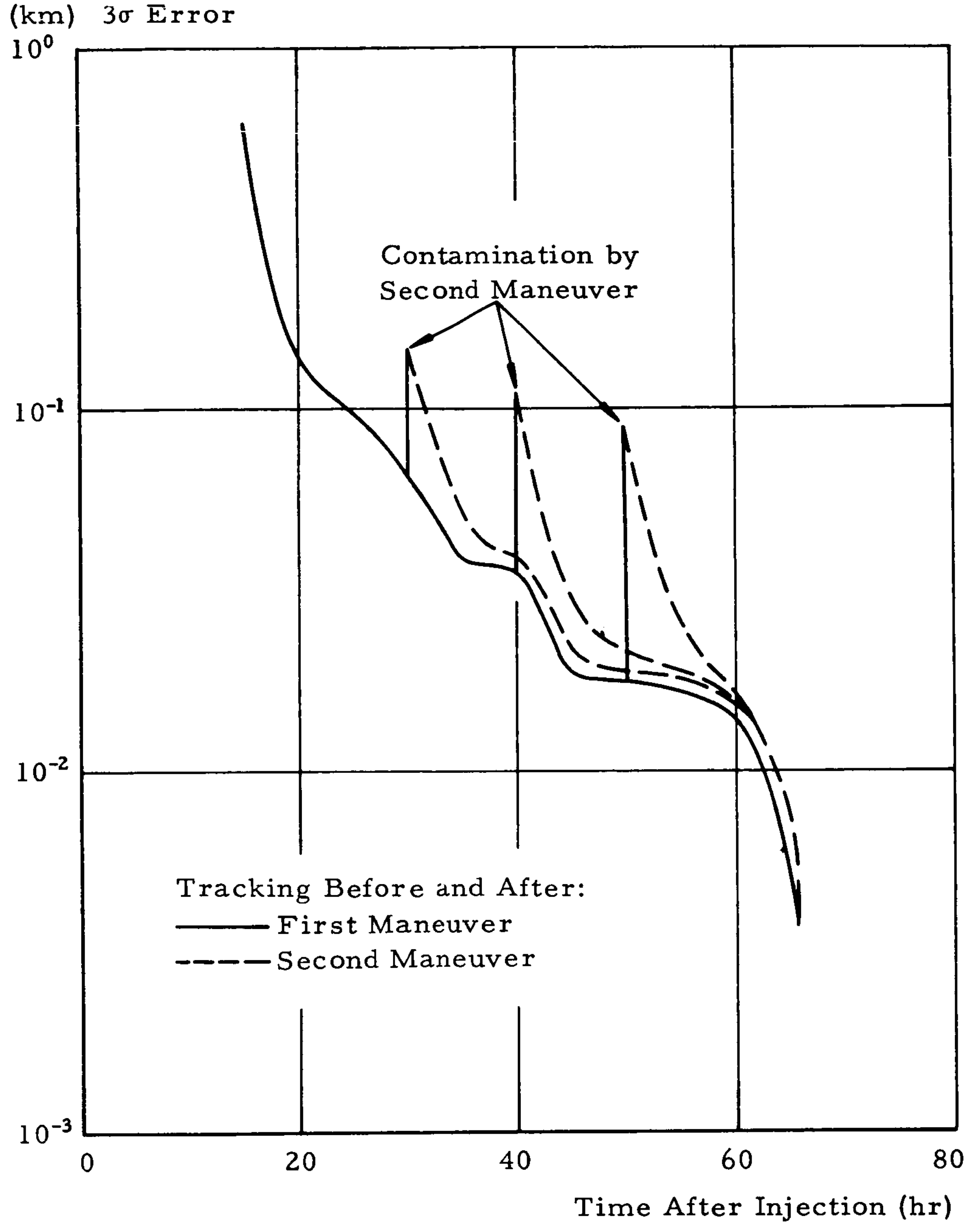


FIG. 21 CLOSE APPROACH DISTANCE ERROR AFTER SECOND MANEUVER

by maneuver performance at 50 hr rather than 30 hr. However, the absolute level is sufficiently low that the factor is not significant.

A third consideration is the accuracy of redetermination of the transit flight path with tracking data after the maneuver. The accuracy of this redetermination determines the accuracy with which the orbit braking maneuver can be commanded. The parameters of interest in this regard are the uncertainty in state variables at the predicted time of lunar close approach, since the braking maneuver is assumed to take place at this commanded time. The accuracy of this determination, assuming tracking until five hours before close approach but varying time of second maneuver, is shown as the last two entries in Table X.

The error in altitude prediction at a fixed time is somewhat larger than for the close approach event. The errors in orbit determination (neglecting systematic errors), if no second midcourse maneuver and if no midcourse maneuver at all were performed, are also shown for comparison. The final orbit determination for the braking maneuver is about twice as accurate if the second midcourse maneuver is performed at 30 hr rather than 50 hr. While this factor is not unusually significant, an early maneuver appears desirable because of the requirement for orbit redetermination before the braking maneuver.

G. EFFECTS OF SYSTEMATIC ERRORS IN ORBIT DETERMINATION

As shown previously in Section II, systematic errors limit the accuracy with which the earth-moon transit trajectory can be determined. The influence of the systematic error sources upon the midcourse maneuvers will be to increase the ΔV required and to alter to some extent the optimum times for performance of maneuvers. The analysis of these effects is incomplete at this time. However, some estimates can be given.

There are two classes of systematic errors to be considered. One class includes such errors as in the earth gravitation constant (GM) which will bias the preflight trajectory calculations and result in an incorrect flight path even if there were no injection guidance errors. The effect of these errors will be at least partially detected by tracking, whether the error sources are included as unknowns in the orbit determination or not. If they are not included as unknowns, the observed deviation of the actual flight path from the predicted will be attributed to the other parameters solved for in the orbit determination. There is a difference in the accuracy with which the orbit determination can be performed if the systematic errors in this category can be solved for, as shown in Section II. In either case, however, the additional ΔV requirement is about the same.

TABLE X

3σ ERRORS IN PREDICTED STATE VARIABLES AT PREDICTED TIME OF CLOSE APPROACH

DSIF Tracking Assumed With:	V (m/s)	α (deg)	ϵ (deg)	R (km)	ψ (deg)	λ (deg)
No Midcourse Maneuver, Systematic Errors	3	0.30	0.60	8	0.42	1.00
No Midcourse Maneuver, Solve for Systematic Errors	0.18	1.80×10^{-2}	2.70×10^{-2}	0.57	0.81×10^{-2}	5.10×10^{-2}
No Midcourse Maneuver, No Systematic Errors	0.02	0.10×10^{-2}	0.14×10^{-2}	0.04	0.13×10^{-2}	0.26×10^{-2}
One Midcourse Maneuver (10 hr), No Systematic Errors	0.02	0.14×10^{-2}	0.23×10^{-2}	0.05	0.17×10^{-2}	0.42×10^{-2}
Two Midcourse Maneuvers (10 & 30 hr), No Systematic Errors	0.07	0.17×10^{-2}	0.45×10^{-2}	0.16	0.40×10^{-2}	0.82×10^{-2}
Two Midcourse Maneuvers (10 & 50 hr), No Systematic Errors	0.11	0.17×10^{-2}	0.48×10^{-2}	0.26	0.62×10^{-2}	0.87×10^{-2}

The second class of systematic errors includes those which affect only the tracking measurements, such as station coordinate errors. These errors, if not solved for and eliminated in the orbit determination process, create an apparent but erroneous flight path deviation in the orbit determination. Since the apparent deviation cannot be distinguished from a real deviation, it must be corrected for. This imposes a ΔV requirement as well as creating a true deviation in the flight path. To the extent that the systematic errors in this category can be solved for in the orbit determination and eliminated, they impose no ΔV requirement and contribute no error to the orbit determination.

The total midcourse ΔV requirement for the same systematic errors in both categories investigated in Section II of this report, assuming they are not eliminated before or during the logistics vehicle flight, is of the order of 10 m/s, although this figure varies according to the control scheme and maneuver times assumed.

The effect of the systematic errors upon the timing of midcourse maneuvers has not been fully determined. Preliminary results appear much the same as in Par. III.F, where systematic errors were neglected. However, since the absolute error level of orbit determination is much higher when systematic errors are considered, the significance of the contaminating effect of a midcourse maneuver on the orbit determination will be reduced, permitting a later second maneuver. This may also be desired in order to permit more tracking time for detection of systematic errors.

H. SUMMARY

Although the limited scope of the investigations reported do not really answer the questions posed in the introduction, they do yield some indications and tentative answers.

(1) Two midcourse corrections may be required, the first about 10 hr and the second 30 to 50 hr after injection into the earth-moon transit.

(2) The ΔV requirement for midcourse corrections, the ΔV requirement for correction during the braking maneuver into lunar orbit, and the scheme for midcourse correction are closely interrelated. The requirement is also determined essentially by the magnitude of injection guidance errors. The midcourse ΔV requirement can be reduced at the expense of braking maneuver ΔV and vice versa. An optimization including both maneuvers should be performed.

(3) The midcourse maneuvers cause some reduction (perhaps by a factor of 3) in the terminal accuracy of orbit determination

considering only random observational errors, especially if the second correction is performed late in flight. The importance of this contamination effect on the orbit determination will be reduced when systematic tracking errors are considered, due to the generally higher error level of orbit determination in relation to maneuver accuracy.

IV. LUNAR LOGISTICS MISSION CONTROL

A. INTRODUCTION

The complex mission profile of the unmanned Lunar Logistics System (LLS) requires the capability of command control from the ground, during both normal missions and emergencies.

This requirement is based on the principal desires (a) to utilize well-developed and highly accurate ground tracking systems for spacecraft navigation and (b) to increase the probability of mission success by placing a substantial portion of the decision powers on ground rather than relying on a pure onboard automatic system.

This philosophy closely follows the experience and control concepts established for other NASA programs such as the Integrated Mission Control Center (IMCC) in Houston, Texas, for Gemini and Apollo manned missions, and the Space Flight Operations Facility (SFOF) in Pasadena, California, for unmanned lunar and planetary missions.

This section presents a brief discussion of a tentative LLS mission control concept, the operations which are expected to be performed in support of the LLS mission, and the corresponding requirements which can be fulfilled to a large extent by existing or presently planned facilities. The paper is not the result of a systematic and complete study program. A number of trade-off possibilities exist and optimization studies will have to be performed before a final mission control center concept can be formulated. For this reason, detailed facility and operations plans are not included and development schedules cannot be given.

B. LLS MISSION PROFILE

1. General. A space mission profile is the sequential outline of key events of a space flight operation which are essential for accomplishing the flight mission. For a Lunar Logistics Mission, this profile extends from the earth launch point to the lunar landing site. Mission Control upon the space vehicle is exercised from the earth to assure that the mission profile will be closely adhered to in order to achieve a successful mission.

The Lunar Logistic Vehicle (LLV) under consideration for this discussion is the Saturn V configuration which has the following breakdown:

a. Launch Vehicle

(1) S-IC Stage
(2) S-II Stage
(3) S-IVB Stage

b. Spacecraft

(1) L-I Stage
(2) L-II Stage
(3) Instrument Unit
(4) Payload

2. Profile (Refer to Figure 22).

a. Launch and Earth Orbit (Events 1 through 7, Figure 22)

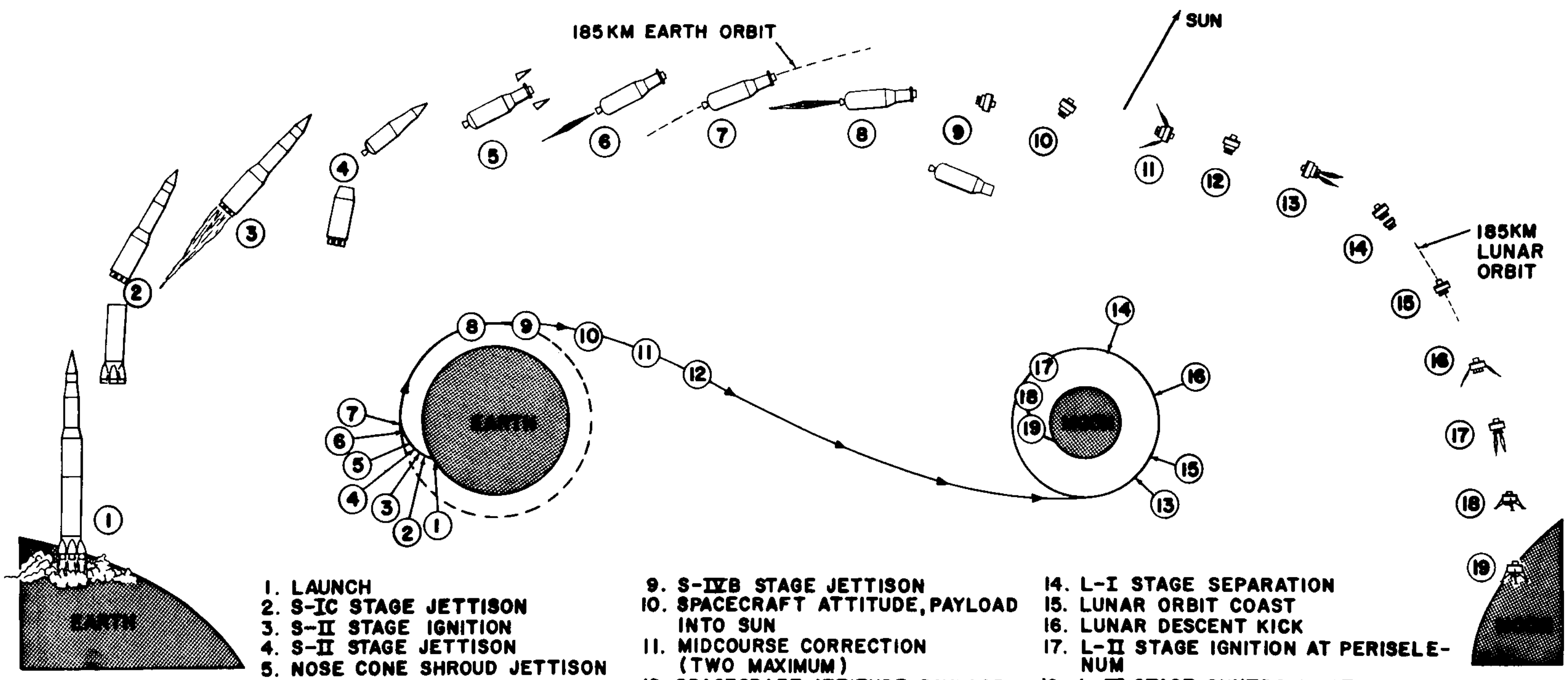
The vehicle is launched from Complex 39 at Cape Canaveral with a variable azimuth (72 to 108 degrees) and boosted by the three-stage launch vehicle into an approximate 185-km circular earth orbit. During the launch ascent, the S-IC and S-II stages are expended and the vehicle nose cone shroud is jettisoned prior to the first ignition of the S-IVB stage. This stage is shut down upon vehicle injection into earth parking orbit.

The plane of the earth parking orbit is determined from optimum flight mechanics conditions for the launch time and date, in accordance with earth-moon flight geometry. The vehicle coasts in this plane from a few degrees to a maximum of one full orbit with S-IVB attitude control, until the departure point for the 72-hour earth-moon transit is reached. Additional coasting orbits might be required under certain circumstances. During this parking phase, the orbit ephemeris is determined by a ground tracking station network. The ephemeris is used to confirm or update guidance data stored within the vehicle inertial navigation system. The S-IVB stage is then re-ignited to furnish the additional velocity required for injection into the transit trajectory.

b. Earth-Moon Transit (Events 8 through 12, Figure 22)

Immediately upon attaining the required earth escape velocity, the S-IVB stage is cut off. This stage may be separated from the spacecraft at this time or delayed to insure observation by the ground tracking network. Its velocity vector is slightly altered to prevent lunar impact. During the lunar transit period (except for midcourse maneuver operations), the spacecraft is oriented with its front end, or payload compartment, into the sun to minimize hydrogen boiloff. The vernier propulsion system of the spacecraft is to be capable of a total velocity change of 100 meters/second, a conservative

SATURN V
LUNAR LOGISTIC SYSTEM
TYPICAL MISSION PROFILE



1. LAUNCH
2. S-IC STAGE JETTISON
3. S-II STAGE IGNITION
4. S-II STAGE JETTISON
5. NOSE CONE SHROUD JETTISON
6. S-IVB STAGE IGNITION
7. INSERTION INTO EARTH PARKING ORBIT
8. S-IVB RESTART AND TRANSLUNAR INJECTION (72 HR. TRANSFER)
9. S-IVB STAGE JETTISON
10. SPACECRAFT ATTITUDE, PAYLOAD INTO SUN
11. MIDCOURSE CORRECTION (TWO MAXIMUM)
12. SPACECRAFT ATTITUDE, PAYLOAD INTO SUN
13. L-I STAGE BRAKING INTO LUNAR ORBIT
14. L-I STAGE SEPARATION
15. LUNAR ORBIT COAST
16. LUNAR DESCENT KICK
17. L-II STAGE IGNITION AT PERISELENUM
18. L-II STAGE SHUTDOWN AT APPROX. 3M ALTITUDE
19. LUNAR HOVER AND LANDING

NOTE:
1. THIS DIAGRAM IS NOT IN TRUE PERSPECTIVE OR SCALE AND IS FOR INFORMATIONAL PURPOSES ONLY.
2. COAST PERIOD IS FROM POSITION 8 TO 13.

FIG. 22

estimate for the two envisioned midcourse correction maneuvers (one each approximately 10 and 40 hours after injection into the 72-hour lunar transfer trajectory). Necessary midcourse maneuvers determined by earth tracking of the spacecraft and earth computation are radio-commanded to the vehicle for execution at the proper time.

Approximately four hours before reaching periselenum the inertial platform is realigned by celestial references in order to establish the guidance coordinates relative to the lunar local vertical. Earth-based tracking and computations determine the braking maneuver for placing the spacecraft into an approximate 185-km circular lunar orbit with a plane containing the desired landing site. The braking maneuver command is transmitted to the spacecraft before lunar occultation for execution at the proper time.

c. Lunar Orbit and Landing (Events 13 through 19, Figure 22)

The L-I stage brakes the spacecraft into lunar orbit. This braking maneuver will probably have to take place behind the moon during lunar occultation. Upon completion of braking and emergence from behind the moon, the L-I stage is separated and placed into a lunar orbit that prevents interference with remaining LLS operations. Lunar orbit staytime of the L-II stage with instrument unit and payload compartment is planned to be between two and four orbits. During this period, the vehicle attitude is determined by antenna orientation constraints and minimum hydrogen boiloff requirements. Also, an additional platform realignment to selected stellar references is required.

After lunar orbit ephemeris determination by the earth tracking network, the descent kick maneuver is computed by the Mission Control Center to place the vehicle on a descending orbit with periselenum near the landing site and at about 25 km altitude. Instructions for the main braking to the lunar surface are also computed at this time. The maneuver program is transmitted to the vehicle through the radio command link for execution at the proper time. The descent kick by the vernier propulsion system of the L-II stage will probably not be visible from earth.

At the periselenum of the descending orbit (sensed by onboard equipment) the vehicle is braked by the L-II stage main propulsion system to zero velocity at the landing site (an RF beacon, if applicable) at an altitude of approximately 30 meters. Adaptive inertial guidance with appropriate navigational aids (or TV navigation controlled from earth, if available) assures landing approach accuracy. At an altitude of about 3 meters (sensed by onboard equipment), the L-II stage propulsion system is shut down and the vehicle falls to a stable lunar landing on its landing gear. The payload is then deployed and operated (if applicable) by radio control from earth.

C. OBJECTIVES OF MISSION CONTROL

The mission profile of the unmanned Lunar Logistics Vehicle requires a capability for virtually continuous command control of the space vehicle from a central control point on earth, the Mission Control Center. There are three general control objectives which determine the requirements of a Mission Control Center for the LLV:

1. <u>Flight Path Control</u>. The control center will monitor the entire flight path and will initiate corrective maneuvers if significant deviations from the established mission profile occur.

2. <u>Vehicle Functional Control</u>. The control center will execute all steps of the mission sequence which cannot be performed purely automatically and exercise override of automatic functions when advantageous.

3. <u>Vehicle Malfunction Control</u>. The control center will attempt to achieve optimum usefulness of the mission in case of vehicle malfunctions by corrective actions, including possible modification of the original mission profile.

In manned lunar operations of the Apollo program, the astronaut crew will play a major role in mission control especially during critical operational periods and during emergency situations. In such situations, primary mission control may be temporarily transferred from the Integrated Mission Control Center in Houston to the crew onboard the Apollo command module. Since pilots are not onboard the LLV to perform mission control functions, such functions must originate in the earth-based Lunar Logistics Mission Control Center (LMCC). There is a mandatory requirement for precision tracking of the LLV for reliable reception of telemetered measurements to monitor the vehicle and mission status, and for dependable transmission of control commands to the vehicle. This leads to the following expected main differences between the mission control of manned Apollo missions and unmanned Lunar Logistics missions:

a. As LLV will not have a pilot crew, there will be no interface in operational decisions between the Integrated Mission Control Center and a crew;

b. The LLV will not return to the earth; and

c. The LLV control will probably require more complete ground monitoring of vehicle systems.

It is expected that the total net effect of these three differences will mean a somewhat less difficult task for the mission control of the LLV as compared with a normal Apollo mission control. The differences in the scope of mission control are reflected not only in the quantity and type of onboard guidance control and navigation equipment required but also in the facility, equipment, and manpower requirements of the earth-based mission control complex.

The onboard measuring system must enable the LMCC to constantly monitor the status of all vehicle systems (propulsion, attitude, instrumentation, guidance, control, structural integrity, scheduled operational events, and the internal vehicle environment). During periods when communication to earth is not possible (lunar occultation), data must be recorded for later transmission.

The mission controller will observe vehicle status trends through these onboard measurements, which may indicate an impending equipment malfunction or failure. Within the limitations given by the vehicle system design, by the mission time schedule, and by the usefulness of a degraded mission, some degree of vehicle malfunction control may be exercised by the Mission Control Center.

Monitoring periods of particular importance are during the earth and the moon parking orbits. During these orbital periods, systems checkout of the entire vehicle may be performed before the next major step of the mission profile is initiated. These checkouts would again have the primary purposes of alleviating potential failures and of achieving optimum usefulness of the remaining mission in case of marginal situations or partial malfunctions. An onboard computer, using a predesigned program, will stimulate vehicle subsystems and systems resulting in simple indications of the go-no-go type. In case of a negative checkout result, telemetered vehicle data, vehicle design information, and operational experience factors are used in the Mission Control Center to attempt corrective action and make a decision on whether to continue the flight or modify the mission profile.

Ephemeris determinations of the space vehicle throughout the mission profile must largely be based on earth tracking data. Correlation and comparison of these tracking data with telemetered guidance data can be used to detect guidance system deviations which in turn will be used to update the guidance system coefficients and similar flight control characteristics.

Any power thrust after injection of the vehicle into the earth-moon transit trajectory must be computed in the Mission Control Center and then transmitted by a radio command link to the LLV for execution at the proper time. Examples of this type of function are midcourse maneuvers and braking into lunar orbit.

In addition, there are several mission tasks which have to be performed in a control center for support operations, such as the prediction of acquisition conditions for the tracking station network. Reacquisition of the spacecraft is of particular interest after every period of lunar occultation.

The most difficult and critical phase of a lunar logistics mission is the descent to the lunar surface, especially for the first such operation. If a lunar radio beacon has been emplaced within the desired landing zone by a previously landed spacecraft, the LLV landing sequence is supported by onboard navigation measurements after the beacon has been acquired by the vehicle. If, however, no navigational beacon has been placed on the moon for use of the LLV, a television navigation system would be one possible means to achieve the required landing accuracies and to avoid local landing hazards. This navigation mode presupposes an earth-based navigator within the Mission Control Center who can select a suitable landing site based on TV pictures received in almost real time. The signal transmission delay (about 2.5 sec for the round trip plus operational reaction times) will be considered in navigational computations. Onboard resolvers translate TV camera pointing angles into information for use of the vehicle guidance system during the landing operation.

After the LLV has been landed, the LMCC supervises the remote deployment of the payload, if applicable, and the operation of any equipment which is part of the cargo such as roving vehicle, radio beacon, lunar observatory, or remote TV station. Here again, as in instances of spacecraft malfunctions, the Mission Control Center must be in a position to make control decisions which will assure optimum payload effectivity in case of abnormalities and malfunctions of the deployed payload.

Detailed mission control functions required on a timely basis for the Lunar Logistics Mission profile and the mission control complex will be discussed in Section E, Mission Control Operations.

D. CONCEPT OF LUNAR LOGISTICS MISSION CONTROL

1. Basic Concept. The basic design goal of the Lunar Logistics Mission Control concept proposed at this time is to achieve the three general mission objectives (flight path control, vehicle functional control, and malfunction control) with a minimum addition of new facilities.

Present implementations of other NASA programs show that there are at least three major distinguishable elements of a mission control system:

a. The Mission Control Center;
b. The ground instrumentation network; and
c. A ground communication network

To achieve maximum efficiency of existing tracking stations and communication lines, it appears mandatory for the LLS mission to use existing facilities of this kind on a part-time basis. It is not anticipated that lunar logistics missions will fully saturate the capability of any one tracking and communication network.

It appears also most logical to receive all required intelligence (tracking and telemetry) through one focal point, an instrumentation network and communications control center. Using an existing facility for this function would avoid the problem of coordinating a complex ground network on a periodic and part-time basis in the Mission Control Center.

The Mission Control Center would retain in this concept only the computing, analyzing and decision-making functions. These functions require such intimate knowledge of the space vehicle design that a physical separation from the development center in charge of the LLS must be considered highly impractical. It is therefore proposed to place this control center facility in immediate proximity of the development center. In deviation from the Apollo Integrated Mission Control Center approach, this control center would not communicate to the remote sites directly but through a network control center (such as Goddard Space Flight Center, GSFC), which would have the experience and capacity to manage similar tasks for other programs in order to utilize existing NASA tracking and communication facilities to the highest possible degree.

The LMCC would have the capability to exercise all aspects of Mission Control. Its essential internal elements are the communications terminal, a computer complex, a data and mission status display system, and a central control organization.

The support to enable the LMCC to fulfill all its functions is furnished over a world-wide communications network which extends from the Mission Control Center through the ground network control center to the instrumentation stations. Over the global communication network, all data, voice, television, facsimile, and teletype messages flow between the Mission Control Center through the network control center to all remote instrumentation stations. The communications network control center manages this network and serves as communications switching center. The instrumentation network control center manages and coordinates the operation of the network of tracking and telemetry receiving stations. It is proposed that both instrumentation and communication network control centers are physically united within one common facility, a Ground Instrumentation Control Center (GICC).

Because of the fundamental difference in equipment configuration and operating techniques between stations supporting near-earth space flight operations and those used for lunar and planetary flight operations, the ground instrumentation network has two distinct segments: (1) the near-earth sites, and (2) the deep-space stations. Stations of both segments are capable of space vehicle tracking, of telemetry data reception from the spacecraft, and of radio command transmission to the vehicle.

An overall schematic of the Logistics Mission Control Concept and its functional support is depicted in Figure 23. In this diagram, the Launch Control Center (LCC) is shown to have an interface with the Logistics Mission Control Center under the concept that mission control is transferred from LCC to LMCC at the point of earth orbit injection of the LLV.

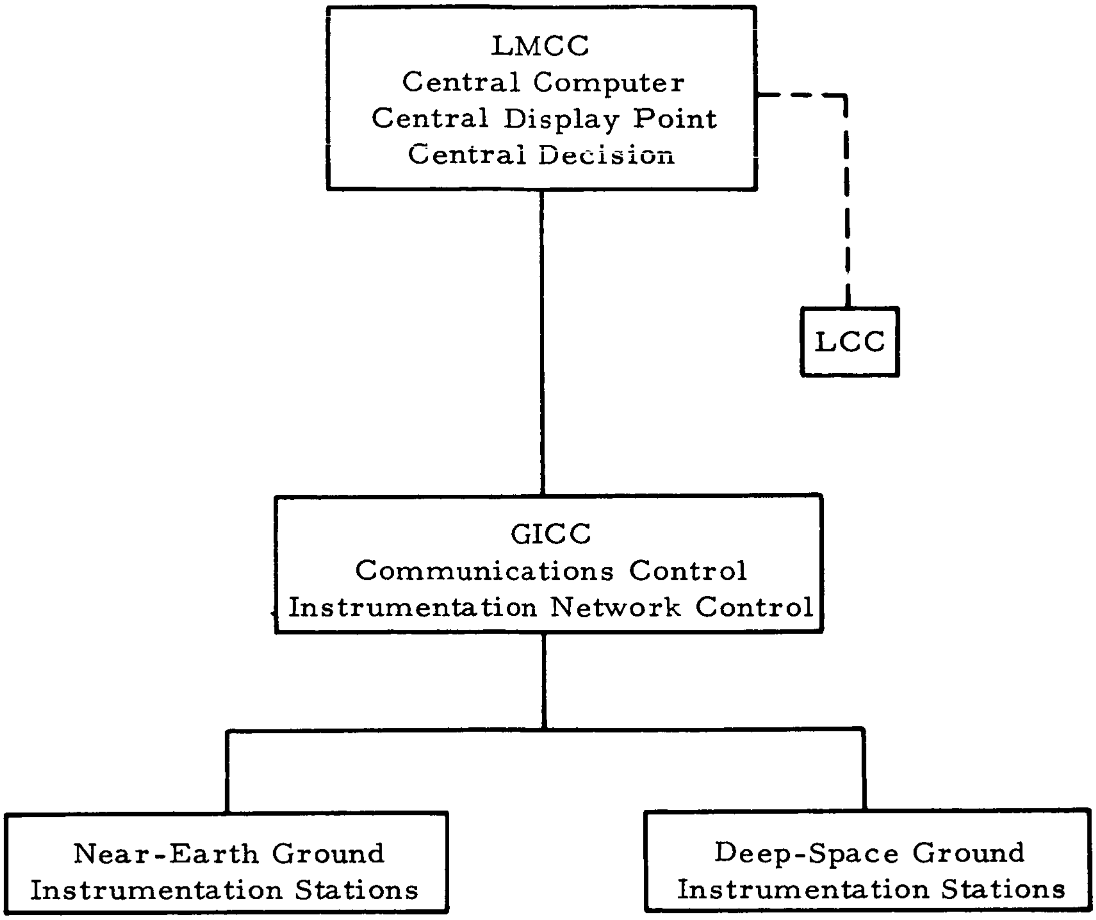
The flight phase from launch to earth orbit injection is assumed to be the responsibility of the LCC. The LMCC monitors this phase in order to be aware of the launch and earth ascent operations status and to gain data that will assist in the proper interpretation of operational data obtained subsequently during the orbital flight.

2. Mission Control Center.

a. Adaptability of Existing NASA Mission Control Centers

Two NASA space mission control centers whose general functions for their presently assigned projects are in several aspects similar to those of a LMCC are now in design and implementation stages. These two centers are the Space Flight Operations Facility of the Jet Propulsion Laboratory (JPL), at Pasadena, California, and IMCC of the Manned Space Center (MSC), at Houston, Texas.

The SFOF is primarily designed to support JPL Lunar and Planetary Scientific Research and Exploration Flight Programs, even though some support of other NASA space missions not directed by JPL is included in the present JPL planning. One of the important stipulations is that the non-JPL user of the SFOF adopt the operational procedures developed by JPL. This may not always be compatible with the functional control requirements of a particular non-JPL space project. It is specifically anticipated that because of the predominantly scientific nature of the JPL missions some of the SFOF



Legend:

——— Direct Support
– – – – Interface
LCC - Launch Control Center
LMCC - Logistics Mission Control Center
GICC - Ground Instrumentation Network Control

FIG. 23 LOGISTICS MISSION CONTROL CONCEPT elements are not easily adaptable for Lunar Logistics Mission Control operations, and that facility modifications, physical expansion, and augmentation for support of LLS missions would be required to a large extent. Furthermore, a review of the current JPL Lunar and Planetary Flight Programs schedule indicates that SFOF will not be able to support non-JPL space flight projects on a large scale.

The IMCC in Houston, Texas has been specifically planned to support all operational phases of the Gemini and Apollo Programs including astronaut training, mission control systems checkout activities, mission control operations, and post-mission analysis and evaluation. Various IMCC operational elements are specifically designed for monitoring the spacecraft crew and their life-support systems status and for astronaut participation in mission control and operation. Some LLS Mission Control requirements (especially in the lunar descent phase) cannot be supplied by the IMCC without modifications to the present design nor can the IMCC staff be expected to be intimately familiar with all details of the Lunar Logistics System. However, more important than any other considerations for using the IMCC for LLS Mission Control operations is the fact that presently-scheduled Gemini-Apollo activities will tax the dual control capability of this facility fully from its initial operating date in mid-1964 to beyond the end of this decade precluding sufficient time available for Lunar Logistics Mission Control support. Extensive, varied, and lengthy mission control checkouts, exercises, and simulations must be conducted in and from the IMCC in support of manned space flight activities. Several unmanned Gemini and Apollo flights have to be supervised by and evaluated in the IMCC to certify the space vehicle configurations for the subsequent manned missions. Finally, lunar manned flights have to be directed and controlled from the IMCC and then analyzed and evaluated there.

b. Preferred Mission Control Center Siting

It would appear then that neither of the two mission control centers presently in the design and implementation stages can, without major modifications and physical expansion, serve as a Logistics Mission Control Center in addition to the activities associated with their assigned projects. An independent specific mission control center is needed for LLS support. It is logical to propose that the IMCC should be physically located at that NASA field center which will be assigned the prime responsibility for research, development, and improvement of the LLV. Several benefits would accrue from this arrangement:

(1) Technical experts would be available prior to, during, and after lunar logistics mission operations for consultations and timely mission operations analysis.

(2) Better utilization of the LMCC management and operating staff (other than the permanent skeleton staff) would be realized by assigning personnel to other LLS Program tasks at the field center in addition to their mission control operational function. This arrangement would result in enhancing the knowledge and experience of the staff in their assigned LMCC duties.

(3) If the R&D versions of the LLV are built and/or these vehicles are ground tested (alignment, balance, systems checkout and compatibility, captive firing, etc.) at the field center, vehicle systems could be checked for compatibility with the LMCC facility and could be made available for simulation tests.

(4) Shortened communications lines would result by concentrating the major portion of LLS activities at one NASA field center.

3. Ground Instrumentation Network and Control Center. The network consists of distinct segments with an associated control center:

a. The near-earth sites
b. The deep-space sites

Near-earth stations are required for furnishing tracking and vehicle telemetry data to the LMCC for the flight operations from earth ascent through space vehicle injection into the earth-moon transit trajectory. Selected stations must also be capable of relaying commands to the LLV. This capability exists in the 17-station Mercury network which is now being augmented to provide PCM telemetry reception and digital command capability for the Gemini and Apollo programs. Later improvements will include on-site data processors, high-speed data transmission circuits, and unified S-band systems for primary stations.

The stations of the Gemini-Apollo network would be depended upon to furnish coverage during the near-earth phases of LLS missions. Only minor additions of on-site equipment and operating personnel would be required for this support. Because of the normal limit of one earth orbit in the LLS mission profile, the support required for lunar logistics operations from these stations will be considerably less than that for Apollo missions (approximately three earth orbits) and no critical interference from concurrent LLS and Apollo operations is anticipated.

The deep-space stations similarly furnish tracking and vehicle telemetry data to the LMCC for operations from space vehicle injection into the earth-moon transit through mission completion. All deep-space stations are capable of relaying commands to the LLV. The existing deep-space stations form the Deep Space Instrumentation Facility (DSIF) of the Jet Propulsion Laboratory. Its three station locations (Goldstone, California; Johannesburg, South Africa; and Woomera, Australia), with 85-foot diameter transmitting and receiving antennas, are spaced approximately 120 degrees apart in longitude around the earth. The DSIF workload will increase for the remainder of this decade because of the heavy JPL Lunar and Planetary Scientific Flight Program. For this reason, a second set of three DSIF-type stations is planned for the unmanned space programs. In addition, another deep-space ground instrumentation network consisting of three stations approximately 120 degrees apart in longitude is planned as primary support for the Apollo program. The new stations anticipated in the South-Central United States, at Canberra, Australia and in Southern Europe (Spain or Sicily) will also have 85-foot diameter antennas and unified S-band capability (PCM telemetry, angles, range and range rate, television, and command). The DSIF stations for the unmanned programs will serve as backup to the primary Apollo deep-space instrumentation network or the JPL DSIF network on a least-interference basis with the other space programs.

In the analysis of tracking station availabilities, Reference 4 was used as an authority.

Indications are that the IMCC in Houston, Texas, will control and coordinate the near-earth and deep-space ground instrumentation network supporting the Gemini and Apollo programs and will perform all computations to accomplish this. This is a departure from the operating philosophy used by the MSC for the Mercury program, for which GSFC is the Instrumentation Network Control Center. For most efficient utilization of existing facilities, the Mercury program philosophy of using GSFC as the ground instrumentation network hub is proposed as the best general compromise for the increasing number of specific space operations. For LMCC support, the GICC would supervise and coordinate both the near-earth and the deep-space segments of the LLS ground instrumentation network. Telemetry data and television information (the latter from a continental U.S. deep-space station) are to be transmitted to the LMCC through the network communications hub. It may be decided later to link the continental United States site directly with the LMCC for rapid two-way data transmission during critical phases of the mission profile.

4. Communication Network. There are four essential levels within the overall LLS communication network:

(1) The Logistics Mission Control Center
(2) The Communication Network Control Center (GICC)
(3) The remote ground instrumentation stations; and
(4) The Lunar Logistics Vehicle

From a communications standpoint, these four levels are essentially in series. Much of the information originates in the LLV and flows to the LMCC, or in the opposite direction.

At the LMCC level are the intra-center communication system for supporting the mission control center functions and the communication terminal where circuits from the GICC and from LMCC interface locations are terminated.

The Communication Network Control Center, the terminal and switching center for the LMCC and remote ground instrumentation station circuits, will manage the communications required during the LLS mission. As an example of existing facilities GSFC now routinely supervises the existing NASA World-wide Communication Network (NASCOM) which is planned to be augmented for more reliable, faster, and wider bandwidth support of NASA space programs. NASCOM by radio, microwave, submarine cable, and landline means is now capable of low and medium bandwidth information transmission. Its augmentation will provide circuit redundancy, more hardwire circuits replacing radio links, and high-speed, wideband capability to overseas instrumentation stations.

At the third level, all near-earth Apollo stations and deep-space instrumentation stations (Apollo and DSIF) proposed for supporting the LLS mission will, through the planned unified S-band system, have the necessary space communication capability with the spacecraft required for Lunar Logistics Mission Control. The modes available through this system are telemetry and television data reception, digital control command transmission and spacecraft verification thereof, and spacecraft information interrogation.

At the fourth communication level, the LLV will have the capability of transmitting telemetry data (direct or tape recorder playback after lunar occultation) and interrogation-response information to the ground instrumentation stations and of receiving control commands from earth.

E. MISSION CONTROL OPERATIONS

1. Mission Control Actions. The role of a ground control complex in operating an unmanned vehicle must be carefully defined. The mission profile and event sequence must be analyzed to determine those tasks better performed automatically within the vehicle and those tasks where ground support is necessary or preferable. A tentative list of actions to be possibly assigned to the mission control complex is presented in Table XI. An exact action list can only follow complete definition of the vehicle and mission profile, but Table XI will serve to indicate the magnitude of support required from the mission control complex.

The ground control actions presented are categorized as primary or secondary. Primary actions are defined as those necessary for accomplishment of the mission and not duplicated onboard the vehicle. Secondary actions are those required for mission success only in a backup capacity in case of malfunction or abnormal conditions onboard the vehicle.

The table is also divided into three flight phases during which the modes of vehicle and/or control complex operation differ significantly. These phases are launch and earth orbit; earth-moon transit; and lunar orbit and landing.

The launch and earth orbit phase includes the mission profile from launch through the second S-IVB burn and up until injection into the earth-moon transit. During the launch into earth orbit (S-IC, S-II, and first S-IVB burn), the vehicle functions completely automatically and the control complex (including the Launch Control Center) only monitors performance.

Once the vehicle is in earth orbit, the principal task of the ground control complex is to perform an orbital checkout of the entire vehicle and correct any errors which have accumulated in the inertial guidance system. The updating of the guidance system is a primary task only for staytime in the earth parking orbit of one hour or more.

During the earth-moon transit, the mission control complex must perform the essential orbit determination and navigation function, instructing the vehicle in the performance of vernier midcourse maneuvers and the braking maneuver into the lunar parking orbit.

Finally, during the lunar orbit and landing phase, the mission control complex must track, determine the vehicle orbit, and direct the onboard guidance system to the landing site.

Table XI Mission Control Actions: Phase I - Launch and Earth Orbit

Event	Mission Control Actions: Primary	Mission Control Actions: Secondary
S-I Burn		Monitor space vehicle status: This includes monitoring of all systems capability, vehicle performance, guidance, H_2 pressure, payload condition, and function of onboard computer-controlled events.
S-II Burn		Monitor space vehicle status. Verify shroud separation.
S-IVB First Burn		Monitor space vehicle status
S-IVB C.O.	Determine injection conditions Predict station acquisition conditions	Monitor space vehicle status
Parking Orbit	Command H_2 venting data playback (if applicable) Determine H_2 venting impulses (if applicable) Determine ephemeris Predict station acquisition conditions Compute escape burn Orbital vehicle systems checkout: includes guidance updating, time synchronization	Monitor space vehicle status: H_2 pressure and accelerometer readings should be recorded for command playback to determine H_2 venting impulses.
	Transmit escape burn command Start TM recorder to record S-IVB escape burn performance in event of unobserved burn.	Verify command reception
S-IVB 2nd Burn		Monitor burn performance if visible.

Table XI Mission Control Action Phase II - Earth-Moon Transit

Event	Mission Control Actions: Primary	Mission Control Actions: Secondary
S-IVB 2nd Burn		Monitor burn performance and space vehicle status if visible.
Earth-Moon Transit	Predict acquisition conditions	
	Acquisition (assuming unobserved S-IVB Burn)	Monitor spacecraft status: This includes vehicle orientation.
	Command recorder playback	Evaluate S-IVB burn performance
	Command S-IVB separation (if not performed automatically)	Verify separation
	Command high-gain antenna deployment	Shutdown unnecessary and malfunctioning equipment
	Command RF systems adjustments	
	Determine ephemeris	
	Compute first midcourse correction maneuver	
	Command first midcourse correction	Verify command reception
First Maneuver		Monitor command execution
	Determine maneuver impulse	Monitor spacecraft status
	Determine ephemeris	
	Update ephemeris	
	Compute second midcourse correction maneuver	
	Update guidance	
	Transmit second midcourse correction	Verify command reception
Second Maneuver		Monitor command execution
	Determine maneuver impulse	Monitor spacecraft status
	Determine ephemeris	
	Update ephemeris	
	Compute lunar braking maneuver	
	Transmit braking maneuver	Verify command reception

Table XI

Event	Mission Control Actions	
	Primary	Secondary
	Transmit braking command Update guidance Adjust RF systems: change to omni-directional acquisition antenna	
Lunar Approach	Initiate approach sequence: turn on altimeter, start TM Recorder, etc.	
Occultation	Predict earth station acquisition (extreme cases: perfect maneuver/ no maneuver)	
L-I Burn		

Table XI Mission Control Action: Phase III - Lunar Orbit And Landing

Event	Mission Control Actions: Primary	Mission Control Actions: Secondary
L-I Burn		
Lunar Orbit	Reacquisition: including RF systems adjustment	
	Command TM recorder playback	Command L-I separation if necessary
	Determine ephemeris	Determine L-I performance
	Synchronize timer	Monitor spacecraft status
	Command lunar beacon search (if applicable)	Compute alternate landing sites
	Command occultation sequence: RF systems adjustment, start TM recorder	
Occultation	Predict acquisition conditions	
	Reacquisition	
	Command TM recorder playback	Monitor spacecraft status
	Determine ephemeris	
	Command lunar beacon search	
	Compute descent command	
	Transmit descent command	Verify command reception
	Orbital vehicle systems checkout	
	Command occultation sequence	
Occultation	Predict acquisition conditions	
Hohmann Maneuver		

Table XI

Mission Control Actions

Event	Primary	Secondary
Transfer Ellipse	Reacquisition Command TM recorder playback Determine ephemeris Command beacon search	Check maneuver execution Backup descent commands Monitor spacecraft status
L-II Burn	Command TV navigation (if applicable)	Monitor L-II performance
Landing	Determine landing point Deploy payload	Refine trajectory Evaluate overall space vehicle performance
Payload Operations	Direct payload control or transfer payload to sponsor	Monitor payload status

During all three phases, vehicle performance is monitored in order to detect and, if possible, correct abnormal conditions, increasing the probability of mission success.

2. Division of Responsibilities Between LMCC and GICC. As described in Par. IV.D.1, it is proposed to make use of existing ground tracking stations coordinated by a Ground Instrumentation Control Center. The GICC would be responsible for those tasks required specifically for operation and control of the ground instrumentation network. The Logistics Mission Control Center would retain responsibility for mission control and would perform those tasks related to operation of the vehicle. The relationship of the LMCC and GICC would be similar to that existing between the range user and the Atlantic Missile Range.

The principal responsibilities assumed by the LMCC, the GICC, and the tracking stations are summarized in Table XII.

The flow of information during a typical action sequence is shown in Figure 24. This sequence represents the steps necessary for updating of the guidance system while in the earth parking orbit. Other action sequences (e.g., vehicle status monitoring), would be occurring simultaneously with the one shown.

3. Mission Control Center Operations.

a. Task Review.

The operations of the Mission Control Center will fall in two categories, (1) the more or less continuous monitoring and status evaluation tasks, and (2) periodic high priority tasks during critical flight periods.

A portion of the continuous monitoring and status evaluation tasks are associated with the internal control of the ground control complex itself (e.g., communications status). The portion of the continuing tasks associated with the vehicle are primarily concerned with malfunction control. Through monitoring of the vehicle subsystem operation, deviations from predicted performance may be detected, future performance estimated, and action initiated within the command capability to alleviate serious deviations. The periodic tasks are indicated in Table XIII. The items in this table represent primary actions of the LMCC, and are shown in approximate time sequence and scale.

b. Navigational Tasks.

Although vehicle powered maneuvers will be executed under control of the onboard guidance system, it will be necessary to

Table XII

Division of Mission Control Responsibility

Lunar Mission Control Center	Ground Instrumentation Control Center	Tracking Stations
Conducts overall control of mission	Maintains information flow (tracking data, T/M data, television and commands) between LMCC and tracking stations	Maintain communications with vehicle
Performs precision orbit determination for navigational purposes	Exerts operational control of ground instrumentation network	Track vehicle
Evaluates vehicle status through tracking and T/M data	Exerts operational control of ground communications network	Receive T/M and Television from vehicle
Generates command instructions for navigational and functional control of vehicle	Provides acquisition predictions to tracking stations (may require orbit determination)	May perform limited data editing and compression before transmission to GICC
Establishes mission control procedures and conducts preflight preparations (including simulations)		Transmit commands to vehicle upon request of LMCC and verify command reception
		May perform limited data evaluation in orbital checkout and emergency situations

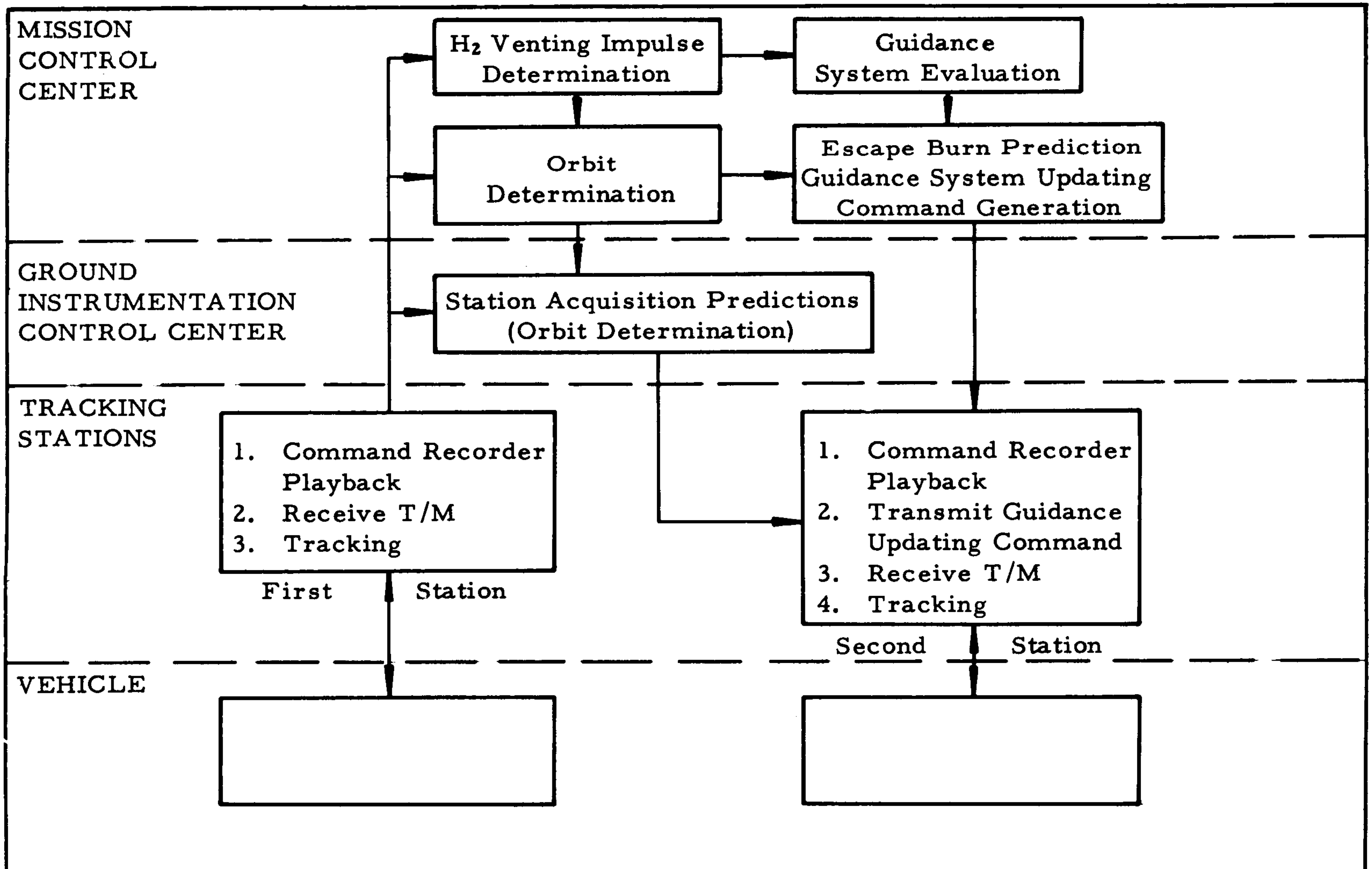


FIG.24 EARTH ORBIT INFORMATION FLOW: GUIDANCE UPDATING SEQUENCE

TABLE XIII TYPICAL TIME SEQUENCE OF MISSION CONTROL CENTER ACTIONS

I. Launch and Earth Orbit

Vehicle Flight Events	Time After Liftoff (min)	Primary LMCC Actions
	0	
S-IC Burn		
S-II Burn		
S-IVB First Burn		
	10	
		Determine Injection Conditions
		Predict Ephemeris
	20	Guidance System Evaluation
		Transmit Guidance Correction
	30	
H_2 Venting Sequence (if applicable)	40	
	50	

NOTE: Brackets indicate approximate times for performance of actions.

Event	Time	Activity
	50	
Australia Tracking (H_2 Venting Data Playback if Applicable)		H_2 Venting Impulse Evaluation (if applicable) Evaluate Guidance System
	60	Determine Ephemeris Compute Escape Burn Update Guidance Orbital Vehicle Checkout
	70	Compute Escape Burn Transmit Guidance Corrections Transmit Escape Burn
	80	
	90	
United States Station Acquisition		Evaluate Guidance System Orbital Vehicle Checkout
S-IVB Second Burn Ignition	100	Update Ephemeris

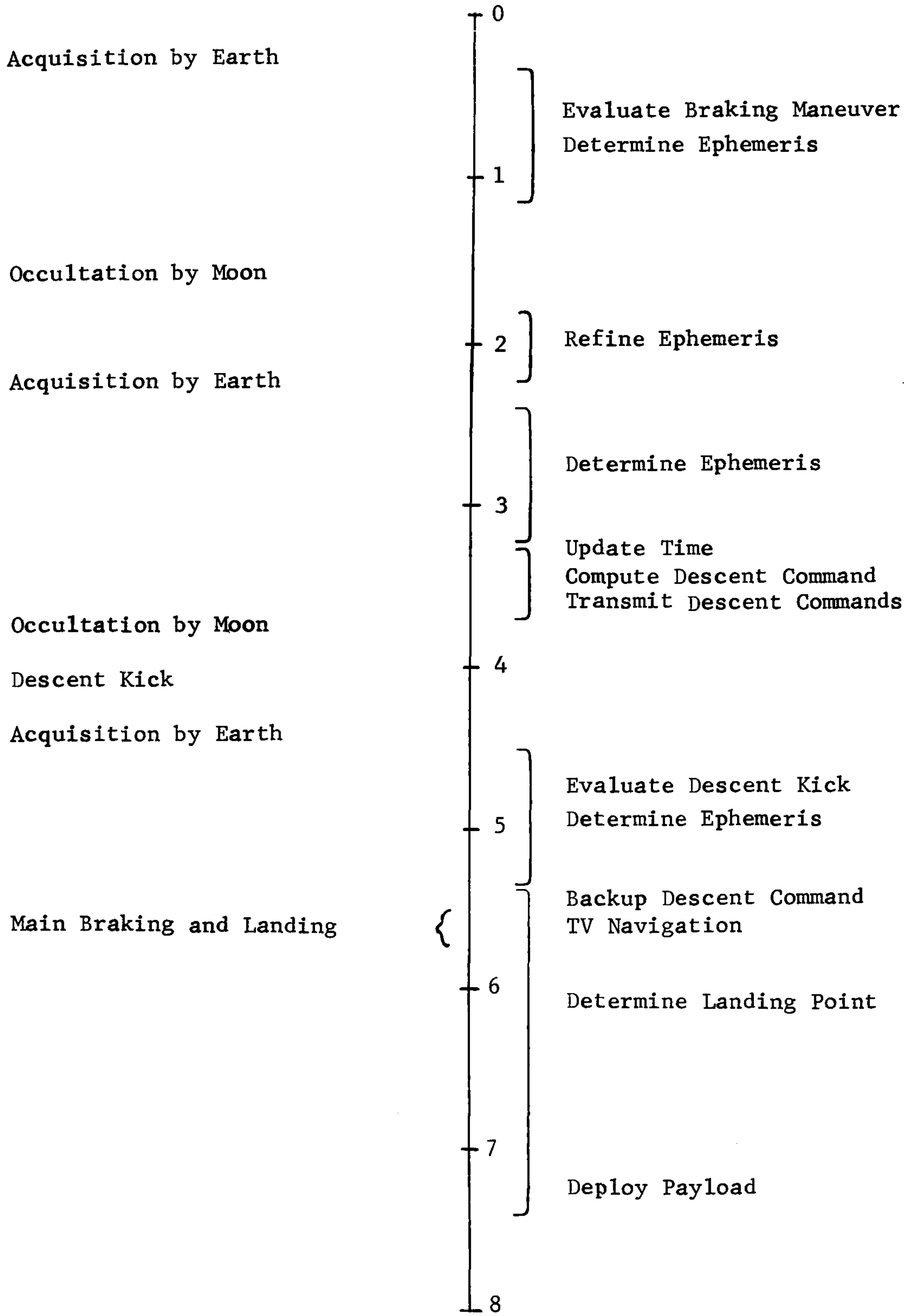
II. Earth-Moon Transit Phase

	Time After Transit Injection (hr)	
S-IVB Second Burn Station #1 Acquisition	0	Determine Burn Performance Determine Injection Conditions Determine Ephemeris
Station #2 Acquisition First Midcourse Correction Station #1 Loss	10	Update Ephemeris Compute and Transmit First Midcourse Maneuver Determine Maneuver Impulse Determine Ephemeris
Station #3 Acquisition Station #2 Loss		
	20	Update Ephemeris
Station #1 Acquisition Station #3 Loss		
	30	Update Ephemeris
Station #2 Acquisition Station #1 Loss Second Midcourse Correction	40	Update Ephemeris Compute and Transmit Second Midcourse Maneuver Determine Maneuver Impulse Determine Ephemeris
Station #3 Acquisition Station #2 Loss		
	50	Update Ephemeris

	50	
Station #1 Acquisition		
Station #3 Loss		
Station #2 Acquisition		
	60	Update Ephemeris
Station #1 Loss		
		Update Ephemeris
Station #3 Acquisition		Update Guidance
		Compute and Transmit L-I Burn Command
Station #2 Loss		
	70	Initiate Approach Sequence
Occultation L-I Braking into Lunar Orbit		

III. Lunar Orbit and Landing

Time After Lunar Orbit Injection (hr)



provide the onboard system with information on which to operate at various stages of the flight. This function may be expected to be exercised five times during the flight.

Guidance System Updating in Earth Parking Orbit

The inertial guidance system is of sufficient accuracy and capacity to permit launch into earth parking orbit, coasting for one full orbital revolution, and execution of the escape burn into the earth-moon transit without information from the ground. However, increased accuracy can optionally be obtained for coast periods of greater than about one-half revolution by updating the guidance accelerometer values with data obtained from earth-based tracking and orbit determination. For coast periods greater than one revolution (if necessary) and in the event of greater-than-predicted guidance errors, updating is required.

The operations to be performed in updating the guidance system might be:

(1) Determine precise orbit by earth tracking;

(2) Predict position and velocity values in guidance system coordinates at a future time, t_o;

(3) Transmit values to vehicle with instructions to begin using these values at time t_1.

The path adaptive guidance system would itself compute the time and program parameters for the S-IVB escape burn, based on the current state variables within it. An alternate approach would be to transmit directly instructions (time, initial state variables, and guidance coefficients) for carrying out the escape burn.

It is anticipated that periodic venting of the hydrogen propellant tanks of the S-IVB may be required in earth orbit. The venting process can be designed to yield little or no net impulse to the vehicle. However, the available energy from this source may be sufficient to warrant the derivation of an intentional impulse, for example, to increase the orbital altitude. Depending upon the degree of control exercised over the time of venting and upon the impulse magnitude, the orbit determination process and the guidance updating could be complicated.

The orbit determination completed by the LMCC at some time t_2 will reflect only the venting impulse effect before an earlier time t_1, which is the end of the last venting sequence about which information has been telemetered to the ground. Thus, the state variable data which can be transmitted to the vehicle from the ground at some still later time t_3 cannot include the effect of venting impulses between t_1 and t_3. The vehicle guidance system must then be prepared to accept information about the correct state variables at time t_1, and predict them forward to time t_3, adding in the effect of the venting impulses between t_1 and t_3 which it measured. The effect of venting may be to place an increased burden upon the onboard guidance computer, as well as complicating the LMCC orbit determination. As an alternative, the final updating computations required to account for the venting between t_1 and t_3 might be assumed by real time ground computation while the vehicle is over the command station, shifting the burden from the onboard computer to the ground network. This would require an on-site computing capacity unless the guidance updating were required to occur over the continental U.S., where high-speed wideband communications to the LMCC would be available.

Earth-Moon Transit Midcourse Maneuvers

Due to errors in injecting the vehicle into the earth-moon transit and uncertainty in physical constants such as the earth gravitation constant, it will be necessary to perform small vernier flight path corrections during the midcourse between earth and moon. Two such maneuvers are anticipated during the flight.

In order to perform these maneuvers, the LMCC must determine a best estimate of the vehicle flight path from tracking and telemetered data, compare this with the desired flight path, and issue commands to correct the path. This process may again be complicated by venting of the cryogenic fuels depending upon venting design, but the problem can more easily be handled during this phase of flight due to continuous communication between the vehicle and ground and the less rigid time frame within which the maneuvers must be accomplished.

Braking Into Lunar Orbit

As the vehicle approaches the moon, its orbit must again be determined. The optimum time and other instructions for igniting the L-I stage and guiding the vehicle into the desired lunar parking orbit must be transmitted to the vehicle.

Descent from Lunar Orbit

Once the lunar parking orbit is established, the LMCC must determine the vehicle ephemeris and update the guidance system in preparation for the descent kick and main braking to the lunar surface.

This operation may be similar to the updating operation performed in the earth parking orbit. The operation will be aided by continuous communications between the vehicle and the ground except during occultation and by the possibility of scheduling the lunar orbit operations within view of the continental U.S.

c. Functional Tasks.

Most functional operations (e.g., turning equipment on and off) will be controlled automatically onboard. However, it will be desirable to initiate certain such actions or action sequences from the ground as a primary mode of control, and to control other actions as an override possibility.

Functional command can be used to accomplish special mission profile requirements, to exercise equipment to test its proper operation, and to accommodate unanticipated conditions in conjunction with malfunction control.

A list of a few functional control actions of particular interest in the Lunar Logistics Vehicle profile is given below:

(1) Turn the onboard T/M recorder on before periods when ground T/M reception is not possible. Command recorder playback when required.

(2) Initiate programmed sequences of actions required on the vehicle during particular flight phases. Such actions may include switching RF power levels, and switching between high and low gain antennas during periods when vehicle attitude must be changed or ground track loss and reacquisition must occur.

(3) Command vehicle events such as S-IVB and L-I separation. If the escape burn of the S-IVB is not visible to the ground instrumentation network, it may be desirable to delay the S-IVB separation until a time when the vehicle may be observed from the ground.

(4) Control over timing and direction of possible H_2 venting impulses may be exercised in order to make use of them for flight path control.

d. Malfunction Control.

The term malfunction control is intended to imply action to accomplish maximum mission success in the presence of unanticipated environmental conditions or equipment performance.

The logistics vehicle will be designed to operate automatically over a range of possible conditions, flight performance deviations, and equipment performance variations in order to insure a high confidence in mission success. This basic design flexibility, in combination with the analysis and computation capability of a ground control center, can permit mission success in the presence of some unanticipated conditions. A careful trade off of reliability, cost, onboard weight, and related factors must be conducted in each specific case.

The elements required to exert malfunction control are (1) measurement and transmission to the ground of critical data necessary to evaluate vehicle performance; (2) a ground facility equipped to analyze the vehicle performance, detect abnormal conditions, and predict future performance; and (3) the capability of actions to correct the abnormal conditions or permit the accomplishment of the mission in spite of them. The argument may be made that anticipated problems should be eliminated by proper design, and unanticipated problems will either not be sufficiently instrumented or not be susceptible to control action. However, no design is perfect; sufficiently flexible and complete instrumentation must and can be provided to permit post-flight detection of reasons for vehicle failures; and a flexibly-designed system will permit control or neutralization of many malfunctions. Recent experience in several programs as well as recent analyses performed (Ref. 5) support this conclusion. The cost of implementing malfunction control must, of course, be weighed against its value.

Within a scope which must be defined by analysis, the LMCC must be capable of receiving and reducing telemetered vehicle performance data; evaluating the vehicle status; predicting future performance; and formulating and commanding effective control actions.

F. REQUIREMENTS

1. Ground Instrumentation.

Near-Earth Network

Ground instrumentation requirements during the launch phase of the mission will essentially be satisfied by the Atlantic Missile Range, since they duplicate the requirements for other Saturn V launches. Requirements in the earth parking orbit are similar to those for manned Apollo flights, but are reduced in magnitude.

The existing and programmed stations of the Apollo near-earth network and some additional stations which should be available for use are shown in Figure 25 and listed with pertinent characteristic

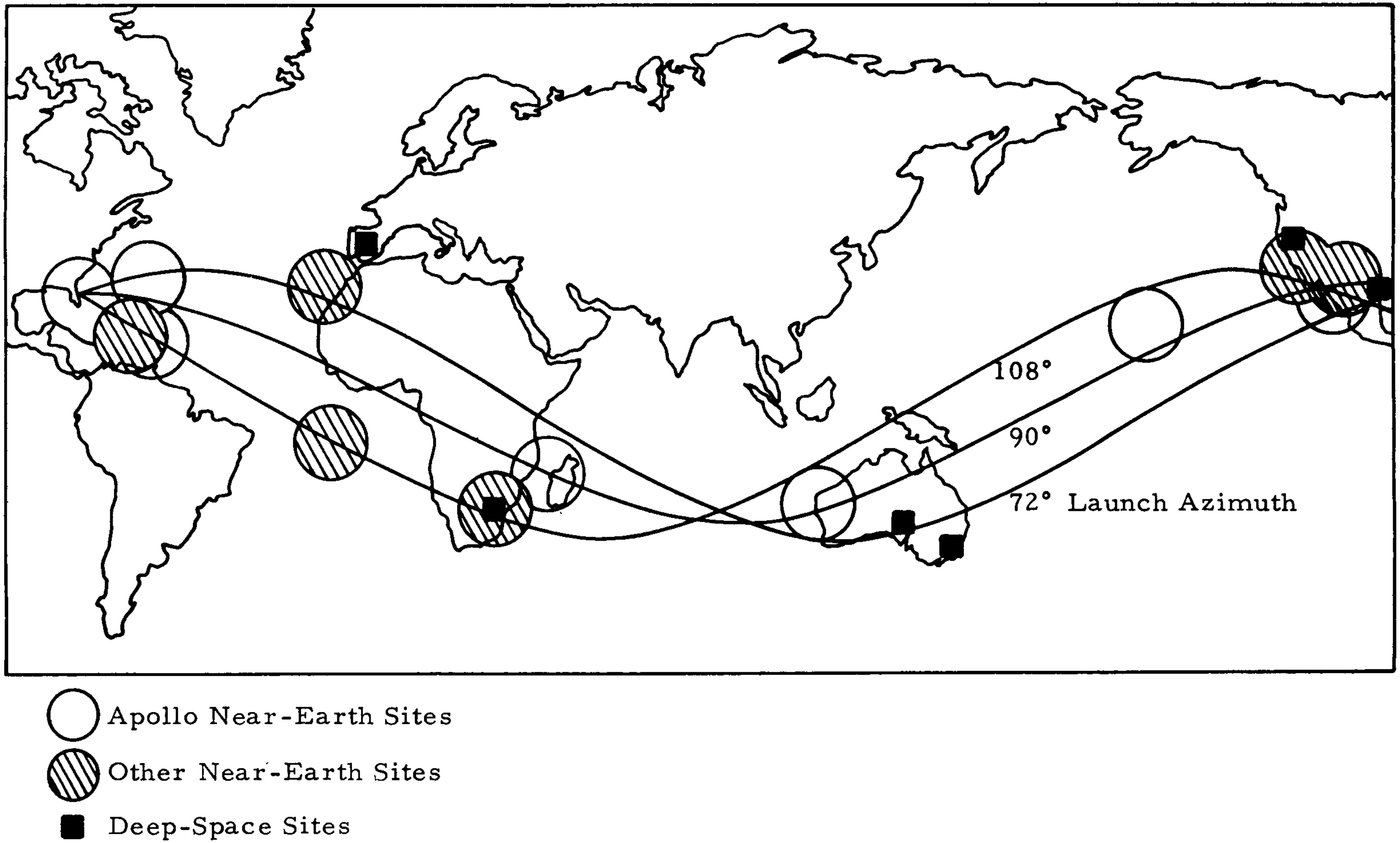


FIG. 25 PROJECTED GROUND INSTRUMENTATION NETWORK SITES

data in Table XIV. Since the Apollo net will be equipped for orbital checkout of and communications to the S-IVB stage, its use is preferable. However, other sites could also be used. The minimum basic requirements for earth orbital coverage of the logistics vehicle is tracking, telemetry, and command capability twice per orbital revolution for the first and possibly three revolutions. A range of launch azimuth from 72 to 108 deg must be considered. Existing and planned sites appear sufficient.

Observation of the S-IVB escape burn is considered highly desirable but not essential, provided tracking shortly before and after the burn can be provided. Essential telemetry data can be recorded onboard for later transmission to the ground. However, if possible, in the presence of other launch constraints, the escape burn may be scheduled over existing stations or ships.

Deep-Space Network

The Deep-Space Network will track the vehicle from shortly after vehicle injection into the earth-moon transit through the remainder of the mission. For this task three stations separated by approximately 120 deg in longitude are required. Table XV shows the existing and planned deep-space stations with 85 foot antennas which would be applicable to the logistics mission (also see Figure 25). All of these stations will be operating in the required time frame and will be similarly equipped. It is believed these stations should be capable of handling the low-firing-density logistics mission in addition to their other programs.

The stations will all use a unified S-band system providing angle, range, and range rate tracking data as well as telemetry, television, and command communications.

The precise stations to be used for the logistics mission are not critical so long as one from each geographical area is used. Selection of stations should be based on projected work loads and station capabilities. However, the station chosen in the continental United States will be a prime station and should be closely and directly linked to the Mission Control Center. This will be necessary for rapid communication with the vehicle during critical portions of the mission profile. The entire sequence of operations in lunar orbit for example, might be scheduled to occur within view of this prime station. Because of the need for wideband TV quality communication from the spacecraft to the primary station, the cost of communications must be traded against the cost of a station near the Mission Control Center. It may be expected that the workload of the continental U.S. stations will be higher than for the overseas stations due to research and development

Table XIV

Projected Near-Earth Station Capabilities

Station	Equipment	Status	
I. Prime Apollo			
Antigua	C,T,U	Planned 1965	AMR
Bermuda	C,T,U	Planned 1965	Apollo
Cape Canaveral	C,T,U	Planned 1965	AMR
Carnarvon	C,T,U	Planned 1965	Apollo
Guaymas	U	Planned 1965	Apollo
Hawaii	C,T,U	Planned 1965	PMR
Madagascar Ship	U	Planned 1965	Apollo
San Salvador	C,T,U	Planned 1965	AMR
Ships (3 for Apollo)	U	Planned 1965	Apollo
II. Other Stations			
Ascension	C,T	Operational	AMR
Canary Isles	T	Operational	Mercury
Point Arguello	C,T	Operational	PMR
Puerto Rico	C,T	Operational	AMR
South Africa	C,T	Operational	AMR
White Sands	C,T	Operational	WSMR

Equipment Code:

C C-band radar
T Telemetry
U Unified S-Band

Table XV

Deep-Space Stations (85 Foot Antennas)

A. Continental United States			
1. Goldstone, California	2	Existing	Lunar and Planetary Program + Apollo Backup
2. South Central United States	1	Planned	Apollo Primary
B. Africa-Europe			
1. South Africa	1	Existing	Lunar and Planetary Program
2. Southern Europe	2	Planned	Apollo Primary, Lunar and Planetary Program + Apollo Backup
C. Australia	3	1 Existing 2 Planned	Lunar and Planetary Program Apollo Primary, Lunar and Planetary Program + Apollo Backup

efforts as well as increased use for mission control simulations. Location of an additional station, if required, should be considered jointly with Mission Control Center requirements. The capital expense for a DSIF type station is about $5,000,000 with annual operating expenses of perhaps $850,000. The annual cost of leasing a TV link between, for example, Huntsville and Cape Canaveral (about 1400 km) is estimated at $500,000; the equivalent cost to Rosman, North Carolina, (about 700 km) is estimated at $300,000. The latter costs depend upon distance, availability of existing facilities, and terrain to be crossed.

2. Onboard Instrumentation.

a. Tracking

Near-Earth

During the launch phase, standard electronic and optical tracking aids for Atlantic Missile Range instrumentation will be carried. For the launch and earth orbit injection determination, a C-band beacon will be the primary aid. In earth orbit, beacons may be required for tracking by C-band and unified S-band systems. In addition, horizon sensor and radar altimeter systems will be carried.

Deep-Space

The principal onboard tracking instrumentation utilized during the deep-space portion of the mission (earth-moon transit, lunar orbit and landing) will be a unified S-band transponder for the DSIF ground stations. Additional instrumentation will be utilized during lunar orbit and landing, including possibly a tracking system to be used with a lunar surface beacon, a radar altimeter, and a horizon sensor.

b. Measuring and Telemetry

The measuring program must be carefully planned to permit status evaluation of all critical vehicle systems. The instrumentation of the S-IVB stage will of course be practically identical to that employed in manned Apollo flights. The instrumentation of the Lunar Logistics System spacecraft, in addition to providing thorough information for vehicle status evaluation, must provide primary mission control data in several areas.

Since a large portion of the midcourse and lunar orbit navigation must be performed on the ground, complete information from the onboard navigation equipment must be supplied to the ground. In particular, all impulses received by the vehicle must be accurately measured. A possible problem in this regard will be the measurement of H_2 venting impulses.

The relatively extended time in space with storage of cryogenic fuels will require close monitoring of the vehicle environment.

An onboard recorder will be required for storage of critical measuring data (such as H_2 venting impulses) during time periods in earth and lunar orbit when communications with a ground station cannot be maintained.

Checkout of the S-IVB stage and logistics spacecraft before major periods of thrust application may be required. Checkout of the S-IVB stage would be performed similarly as in manned Apollo flights, and can afford opportunity for experience in this operation as well as increasing the chance for mission success. Checkout would be performed largely through an automatic onboard system, but results would be telemetered to the ground for evaluation.

A television system may be required for monitoring and possible navigation aid during the lunar landing.

3. Communication Network. Considered here are only the general aspects of the communications requirements external to the LMCC. The communication network managed by the Ground Instrumentation Control Center must provide duplex voice and rapid data transmission between the LMCC and the remote ground instrumentation stations. Closed-circuit television transmission must be provided to the deep-space station in the United States.

The information flow required during the Logistics Mission Operation covers a wide spectrum of bandwidth requirements. It is probably convenient to distinguish three categories:

Very wide bandwidth requirement: In this category we find television and telemetry transmission from the vehicle through a remote site and the GICC to the LMCC. Continuous tracking data are transmitted from at least one remote site at a time through GICC to LMCC. The order of magnitude bit rate is 32,000 bits/sec. An additional wideband link is required between LMCC and the Launch Control Center, although this requirement can be limited to a relatively short time after launch. As mentioned before, this link would essentially serve as a one-way information channel to prepare the LMCC for decisions to be made in the subsequent orbital phase. Prior to flight operations, it may also play an important role for data exchange during the launch preparations.

Medium bandwidth requirement: Tracking acquisition data and all network instructions are given from the GICC to the applicable remote sites. In the opposite flow direction, checkout results are received from the vehicle and communicated through remote site and GICC to LMCC. Expected bit rate is in the order of 1,200 bits/sec.

Narrow bandwidth requirement: In this category, there is the exchange of orbit determination results between the LMCC and GICC, the transmission of control commands through the entire chain to the vehicle and finally the retransmission of this command for confirmation. The expected transmission rate is in the order of 400 bits/sec.

The planned or existing NASA world-wide communications network (NASCOM) designed and supervised by GSFC is considered adequate for the medium and small bandwidth requirements. However, it cannot generally satisfy all of the very wide bandwidth nor high speed transmission requirements.

Concerning tracking data some kind of initial data, compression will be required when high volume real time data are necessary. Subsequently, complete (uncompressed) raw data may be transmitted at lower rates.

The several hundred telemetry measurements will also have to be compressed and partially processed at the local remote instrumentation sites before they can be transmitted in real time over the NASCOM net.

Finally, the television pictures require a bandwidth not available within the NASCOM net. However, real time TV transmission is only expected during the final lunar descent maneuver. This can be timed such that reception from the prime continental United States data acquisition site is possible. A TV link to this site will be required.

4. Ground Instrumentation Control Center. It has already been stated that in order to achieve maximum utilization of existing facilities, the concept of utilizing a GICC for support of LLS Mission Control has been proposed.

This control center is expected to exercise complete operational control over the near-space and deep-space ground instrumentation stations around the globe which are proposed for support of the LLS missions, thus relieving the LMCC of this sizeable responsibility. As part of this function the GICC, by use of the associated computer complex, may be expected to perform preliminary orbit determinations in order to furnish timely acquisition predictions to the instrumentation sites. These predictions have to reflect the anticipated effects of any mission control maneuvers to be commanded of the vehicle by the LMCC. Final orbit determinations will be computed at the LMCC to assure current mission control input to vehicle flight operations.

The GICC will also be depended upon to relay raw tracking and all telemetry data and other important mission information from the ground sites to the LMCC and to pass mission control commands from the LMCC to the appropriate instrumentation stations.

5. Mission Control Center.

a. Data Processing and Computation

The heart of the Mission Control Center is its data processing and computation capability. Sufficient capability must exist for reliable accomplishment of the actions described in Par. IV.E.3. Although substantial effort during non-mission periods must be devoted to simulation and training exercises, the anticipated low frequency of logistics flights would permit use of the facilities for pre- and post-flight computation and evaluation. The requirements of the center for substantial real-time vehicle evaluation capability would make it readily adaptable for certain post-flight evaluation activities. A number of principal task categories requiring computational capabilities of varying types may be identified.

(1) Reception, editing, and storage of the large amount of tracking and telemetry data obtained during flight requires large volume, fast access storage capacity with flexible and easily controllable input and output. This facility must be directly tied to external communication lines.

(2) Telemetry evaluation requires facilities for analog-to-digital conversion, analog display of digital data, and fast de-commutation and calibration of measured data. Comparisons with predicted and previous test data must be performed.

(3) Orbit determination requires high-speed and high-precision computation connected with large volume memory for access to tracking data.

(4) Navigational computation requirements are less demanding upon computer memory, but are otherwise similar to (3).

(5) Display generation requires flexible control of specialized input-output devices, as well as access to fairly large storage and communication with other computational elements.

(6) Command generation requires a reliable facility, closely tied to communication lines with provision for closed-loop monitoring to insure proper reception of commands.

There are several possible concepts of satisfying the task requirements. One or more linked large-scale, multipurpose computers can be used. A substantial number of smaller computers can be used making back-up capability easier to achieve, but increasing computer interface problems (see Ref. 5). An intermediate approach would appear to have several advantages. A high-speed large capacity computer of the 7090 class appears to be required for orbit determination and navigation problems. A large computer with more flexible input-output capability and substantial disc storage could be used for data editing and telemetry evaluation purposes. Smaller peripheral computers with communications to the large computers for display and command purposes would afford immediate response to localized demands and more efficient utilization of the large facility. A careful analysis of requirements and trade-off of computer features is required to define a system with high reliability, flexibility, and capacity.

b. Evaluation and Control

For reliable and timely decision and control of the logistics mission, efficient communications to and from the cognizant control personnel must be provided.

A number of principal categories of display and control elements may be identified.

(1) Mission status displays must summarize the overall status of the vehicle and mission control complex, stage of the mission profile, and anticipated actions required.

(2) Displays summarizing the current and predicted performance of vehicle subsystems will be required to provide a basis for control center actions.

(3) Display and command communications for exercise of vehicle functional control are required.

(4) A facility for exercise of navigational control is required to execute guidance system updating, midcourse maneuvers, lunar orbit descent, and possibly terminal landing guidance by remote television control.

(5) A coordinated facility for evaluation and exercise of malfunction control may be required.

(6) A facility for exercise of short or long term control of the logistics payload (such as an automatic roving vehicle) may be required after payload landing and deployment.

(7) Although responsibility for operation of the ground instrumentation network is vested in another center, a display of network status will be required for coordination and overall mission control.

G. CONCLUSIONS

Although the present concept of mission control for the lunar logistics vehicle serves only as a basis for further analysis and not as a definitive result, certain tentative conclusions can be brought forward.

1. A mission control facility tightly tied to the mission and vehicle design, and hence operated by the vehicle developer, is required for reliable and economical vehicle operation.

2. Existing facilities can provide much of the mission control complex. Anticipated ground instrumentation stations appear adequate to perform the logistics mission in addition to other presently assigned tasks. An assignment of network operation and control tasks to an existing Ground Instrumentation Control Center would permit maximum utilization of existing facilities. Only a moderate scale Mission Control Center would be required as a new facility, and could utilize certain elements anticipated for other missions.

3. Primary actions required of the Mission Control Center include essential navigation tasks such as orbit determination and command of midcourse maneuvers during the earth-moon transit, the braking maneuver into lunar parking orbit, and the descent to lunar landing. Other actions pertain to functional control of specific vehicle operations during special flight events. Examples are control of the data recorder and communications systems during occultation of the vehicle by the moon and deployment and operation of the payload upon landing. Another important category of actions pertains to malfunction control to accommodate for unanticipated or unusual deviations in environment and vehicle performance.

4. Further definition of a mission control concept for the lunar logistics vehicle should be performed in order to firmly establish control requirements and the most efficient and economic means of accomplishment.

APPENDIX

SYMBOLS AND ABBREVIATIONS

A	Semimajor Axis of Orbit
DSIF	Deep Space Instrumentation Facility
e	Eccentricity of Orbit
GICC	Ground Instrumentation Control Center
GM	Earth Gravitation Constant
GSFC	Goddard Space Flight Center
I	Inclination of Orbit
IMCC	Integrated Mission Control Center
JPL	Jet Propulsion Laboratory
LCC	Launch Control Center
LLS	Lunar Logistic System
LLV	Lunar Logistic Vehicle
LMCC	Logistic Mission Control Center
MSC	Manned Spacecraft Center
MSFC	Marshall Space Flight Center
NASCOM	NASA Global Communications Network
R	Radial Distance From Earth or Moon Center to Vehicle
SFOF	Space Flight Operations Facility
t	Time
V	Velocity Magnitude
α	Azimuth Angle of Velocity Vector
ϵ	Elevation Angle of Velocity Vector
λ	Longitude of Vehicle
λ_N	Longitude of Ascending Node of Orbit
ψ	Latitude of Vehicle

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